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NASA/GODDARD SPACE FLIGHT CENTER

THERMOSPHERE, IONOSPHERE, MESOSPHERE ENERGETICS AND DYNAMICS (TIMED)

THE TIMED MISSION AND SCIENCE PROGRAM

PHASE A STUDY REPORT

OCTOBER 1992

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1.0 INTRODUCTION

A Phase A Study was conducted within the Flight Projects Directorate at the Goddard Space Flight Center to develop a mission concept for the Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics (TIMED) Mission. This mission traces its heritage back to the Atmospheric Explorers two decades ago and the Dynamics Explorer a decade ago. Since 1981, reports from the National Academy of Science, the NASA Advisory Council and recently the Woods Hole Conference of the Space Science and Applications Advisory Committee have been calling for the exploration and quantitative investigation of the mesosphere, lower-thermosphere/ionosphere (MLTI) region. [Reference 1]

TIMED is an exploratory mission designed to carry out a comprehensive investigation of the physical and chemical processes acting within, and upon, the terrestrial mesosphere and lower-thermosphere/ionosphere. To carry out this mission requires two spacecraft: one in a high inclination orbit to give broad latitudinal coverage and one in lower inclination orbit for rapid local time coverage. Remote sensing instruments and in-situ instruments will be flown on each spacecraft. The high inclination spacecraft is called TIMED-H and the low inclination spacecraft is called TIMED-L.

1.1 Background

In December 1990, the TIMED Science Definition Team (SDT) was established by NASA's Space Physics Division to define a satellite program that would conduct research on the energetics, dynamics, and chemistry of the mesosphere and lower thermosphere/ ionosphere. It would be the first of a new "intermediate class" of missions ($150M to $400M) designed to address relatively focused science objectives.

A pre-Phase A Study was conducted by the Advanced Mission Analysis Office (AMAO) during the summer of 1991, to develop the mission concept defined by the Science Definition Team (SDT) [Reference 2]. This mission concept was costed by GSFC's Resource Analysis Office (RAO) and was found to substantially exceed the cost target of $300M (FY 1992 dollars).

In November 1991, the SDT was asked by NASA to define ways to descope their original mission concept. To this end, the SDT developed six alternative mission scenarios which they listed in order of decreasing science and, presumably, decreasing cost. The TIMED Project Team was formed at the time the SDT was starting their deliberation on the descope. The project team worked closely with the science team in the development of the alternative mission scenarios. They also added three more options for a total of nine (see Table 1-1). This close coordination between the science team and the project team was extremely important in reaching a balance between science and cost. The goal was to get the most science for $300M. The engineering team identified the cost drivers and challenged science requirements that were driving the cost. On the other hand, the science team challenged the engineering concepts to make sure that the engineers were selecting the lowest cost methods for implementing the requirements. The exercise of defining the options forced the SDT to rethink what they really wanted. As the engineering studies got underway, the cost drivers became known and real-time adjustments were made to the definition of the options.

1.2 Study Approach

The Phase A Study Team started with a cost target, a strawman set of instruments and a set of mission requirements in the form of alternative mission scenarios that were developed by the SDT, working closely with the project team. These scenarios were ranked in order of

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SUMMARY OF MISSION OPTIONS

FROM SCIENCE DEFINITION TEAM

OPTION 0:	4 YEAR MISSION, 2 SPACECRAFT (23 INSTRUMENTS/15 UNIQUE)
OPTION 1:	REDUCED MISSION TO 2 YEARS (23 INSTRUMENTS/15 UNIQUE)
OPTION 2:	REMOVED 8 INSTRUMENTS INCLUDING BOOM INSTRUMENTS (15 INSTRUMENTS/10 UNIQUE)
OPTION 3:	REMOVED REQUIREMENTS FOR DEEP DIPPING (15 INSTRUMENTS/10 UNIQUE)
OPTION 4:	LARGE SPACECRAFT AT 70 DEGREES INCLINATION (10 INSTRUMENTS/10 UNIQUE) SECOND SPACECRAFT "SMALL EXPLORER" AT 30 DEGREES INCLINATION (3 INSTRUMENTS/NO UNIQUE)
OPTION 5:	ONE SPACECRAFT AT 70 DEGREES INCLINATION - INITIALLY CIRCULAR WITH DEEP DIPPING PHASE (15 INSTRUMENTS/15 UNIQUE)

FROM PROJECT TEAM

OPTION 1C:	SAME AS OPTION 1 EXCEPT USES 5 SMALL "SMEX CLASS" SATELLITES
OPTION 1':	SAME AS OPTION ONE WITH ELIMINATION OF ELLIPTICAL PHASE ON TIMED-L
OPTION 2':	SAME AS OPTION 2 WITH ELIMINATION OF ELLIPTICAL PHASE ON TIMED-L

Table 1-1

decreasing scientific return to aid the project team in getting the mission cost below the cost target. The study approach was to develop engineering concepts for each mission, cost each mission and then make a selection of a single mission for additional study and further cost analysis. The selected mission would then be taken forward to Phase B, where the actual selected instruments would replace the strawman instrument set.

1.2.1 Design Drivers

The science requirements which drove the spacecraft design for the study were the instruments themselves, the low perigee ("dipping") and the varying orbit profiles. These drivers are obviously not all independent. The instruments are what drive the orbit profiles and, in particular, the low perigees. However, unique considerations in each of these drivers justify singling them out.

Since missions are flown for the instruments, accommodation of the instruments is, and should be, the most important design driver on any mission. However, on TIMED there were some unique characteristics which had to be considered. Each of the TIMED spacecraft must fly a mixture of remote and in-situ instruments. By their nature, these two types of instruments have conflicting requirements. Remote instruments prefer circular orbits and in-situ instruments prefer elliptical orbits with low perigees.

Another difference is that in-situ instruments prefer to be velocity oriented facing the "ram" direction and the remote instruments prefer to be nadir oriented. Clearly, in an elliptical orbit, if one face of the spacecraft is maintained in the nadir direction, the velocity vector will not remain fixed in body coordinates. Similarly, if one face is maintained into the velocity vector, the nadir direction will move with respect to the body. It is impossible to fly an elliptical orbit that will satisfy both conditions simultaneously.

The majority of the strawman instrument complement presented reasonable power, weight and data requirements. However, one instrument, the Infrared Limb Sounder (IRLS), was a true "tall pole" when it came to power. Driven by the power requirement of a mechanical cooler to cool its infrared detectors, the IRLS orbital average power requirement comes in at almost five times that of any other instrument. Even though it was only flying on one spacecraft, the IRLS drove both spacecraft designs because of the study groundrule that the two spacecraft have a common design. However, using ingenuity, the project team accounted for the IRLS power by a minor addition to the spacecraft carrying the IRLS without driving the size and weight of both spacecraft. A trade study was also conducted to investigate the advantages of using a solid cryogenic cooler instead of the mechanical cooler to cool the IRLS. It was decided that the mechanical cooler would be carried during Phase A since it represented the worst case power requirement. This will be revisited again in Phase B after the instrument proposals have been submitted.

The low perigee requirements were clearly a design driver. To sample the atmospheric constituents, the in-situ instruments need to fly through the core region between 60 and 180 km. The atmospheric density at these low altitudes drives the spacecraft design in several ways. First of all, the body must be aerodynamically clean with no deployed appendages. This rules out high gain antennas for communication with the Tracking and Data Relay Satellite System (TDRSS). It also rules out deployed solar arrays. All of the solar cells must be mounted to the body. Once this happens, the power requirement is coupled to the size of the spacecraft. The more power that is required, the bigger the spacecraft becomes. The bigger the spacecraft becomes, the harder the control system must work to control the attitude during the perigee passes. This drives the power requirement up which, in turn, increases the number of solar cells. It is a vicious circle that must be controlled by a careful balance between requirements and design.

The last of the major design drivers are the varying orbit profiles. Because of the mixture of in-situ and remote instruments, the orbits are either elliptical or circular, depending on which instruments are given preference during a particular phase of the mission. The capability to change orbits requires a larger propulsion system than would be needed just to maintain the altitude of a circular orbit. In addition, there is a requirement to occasionally "dip" the perigee of the elliptical orbits down to a lower altitude and then bring it back up. All of these orbital maneuvers drive the size of the propulsion system, which in turn, drives the size and cost of the observatory. Another concern with a propulsion system is the shift in the center-of-gravity as the propellant is used. Control during the deep perigee portion of the orbit requires tight maintenance of the location of the fuel so that the distance between the center-of-gravity and the center-of-pressure can be accurately determined. This leads to a more complex and costly propulsion system.

2.0 STUDY SUMMARY

2.1 Scientific Aims

The primary science objective of the TIMED Mission is to investigate and quantitatively understand the processes responsible for the energy and momentum budgets of the mesosphere, lower-thermosphere/ionosphere (MLTI).

In developing the alternative mission scenarios, the SDT prioritized the science objectives. This was accomplished in several ways. One way was to prioritize by atmospheric regions. No other mission had ever addressed the important coupling between the mesosphere and thermosphere. Therefore, this region was given the highest priority. Next, it was recognized that the coupling between the lower-thermosphere and lower-ionosphere is far less understood than the thermosphere and ionosphere themselves, which have been studied extensively. Hence, this region was given the next priority.

Another approach to prioritization was in terms of science areas: energetics, dynamics, chemistry and ionospheric structure, and electrodynamics. Among these, the SDT agreed that the most important science objective is the energetics and dynamics of the critical transition region between 60 and 180 km.

As a result of the prioritization, the science objectives were ranked in order of priority as follows:

- Determine the temperature, density and wind structure of the MLTI, including the seasonal and latitudinal variations [energetics].

- Determine the relative importance of the various radiative, chemical, electrodynamical and dynamical sources, and sinks of energy for the thermal structure of the MLTI [energetics].

- Determine the sources and magnitudes of gravity waves and planetary waves (extra-tropical and equatorial) in the MLTI region, their spatial and temporal variability and their forcing of the large-scale circulation, and the thermal and constituent structures [dynamics].

- Determine the mean global tidal structures, their annual and height variations, and their effects on the large-scale circulation in the MLTI [dynamics].

- Determine the relative importance of the interactions occuring among the different wave motions (tidal, gravity and planetary) and the extent, causes and consequences of their variability [dynamics].

- Determine the spatial and temporal variability of the radiatively significant minor species and the role that atomic oxygen plays in influencing the radiative balance through collision energy transfer [chemistry].

- Determine the relative importance of the disturbance dynamo, tidal dynamos and magnetospheric penetration fields in determining the global electrodynamics at mid and equatorial latitudes [ionosphere].

- Determine the spatial and temporal variability of the odd-oxygen, odd-hydrogen and odd-nitrogen compounds in the MLTI [chemistry].

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- Determine the roles of neutral winds and electric fields in controlling large and small-scale variability in ionospheric composition and density structures [ionosphere].

2.1.1 Science Requirements

The following science requirement ensure that the mission design will meet the scientific aims and objectives:

- The TIMED Mission must accommodate the strawman instrument requirements.
- TIMED must be a two-observatory mission with TIMED-H (high inclination) at 95 degrees and TIMED-L (low inclination) at 49 degrees.
- In the initial elliptical phase for TIMED-H, the perigee must cover all latitudes from pole to pole.
- One observatory must be capable of flying through the core region and achieve a perigee height equivalent to a density of 6.0×10^{-9} kg/m^3 (approximately 130 km).
- The observatory design life is 2 years with consumables for 4 years.
- The observatory must provide 0.3 degrees attitude knowledge and 1.0 degrees attitude accuracy in all axes.
- The observatory must provide 25 percent instrument duty cycle with worst sun orientation after 2 years.
- Each observatory must carry a mix of in-situ and remote instruments.

2.2 System Concept

The selected mission option is shown in Figures 2-1 and 2-2. It involves two spacecraft flying in orbits with different inclinations to provide the latitudinal and diurnal coverage by the science. The selected option was Option 2' in Table 1-1.

A smaller, lighter spacecraft needs less fuel and is easier to control at low altitudes. Therefore, the goal of systems engineering was to get as small and compact spacecraft as possible and still meet all the instrument requirements. The atmospheric density increases exponentially as the altitude decreases, as shown in Figure 2-3. This meant that the spacecraft had to be aerodynamically clean with body-mounted solar arrays, flush mounted antennas and no booms. The spacecraft had to be carefully balanced so that the center-of-pressure and the center-of-gravity were close together. Also, to keep costs down, both spacecraft had to have virtually identical designs.

The attitude control system provides ±1.0 degrees (3 sigma) pointing in all three axes, except possibly at perigee where a roll error of up to 4.0 degrees is acceptable due to the drag forces. TIMED is a nadir referenced system, which means that the velocity vector (ram direction) will move in spacecraft coordinates for the elliptical orbit. The velocity vector will be lined up with the roll axis at apogee and at perigee.

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TIMED PHASE A MISSION CONCEPT

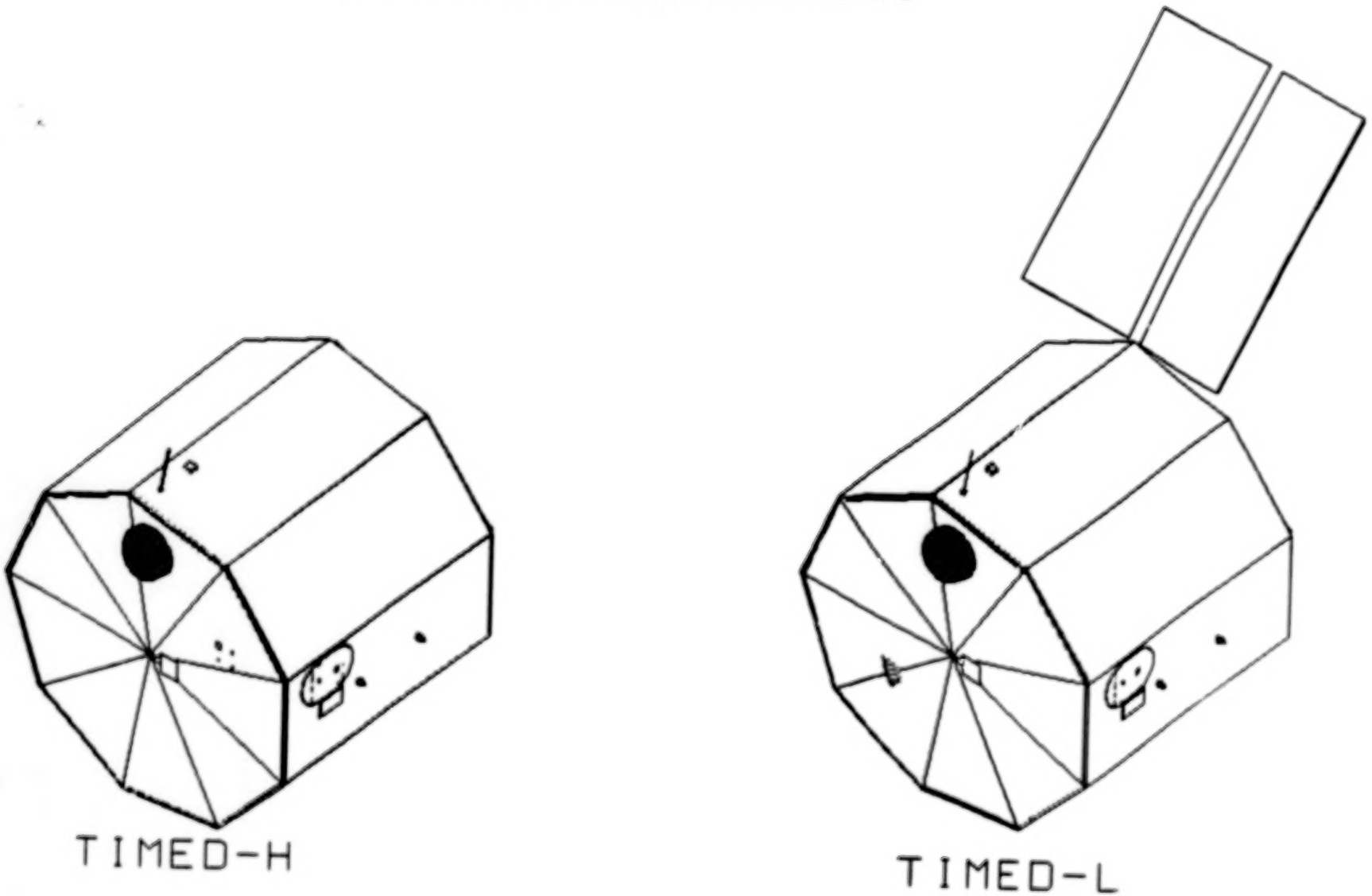
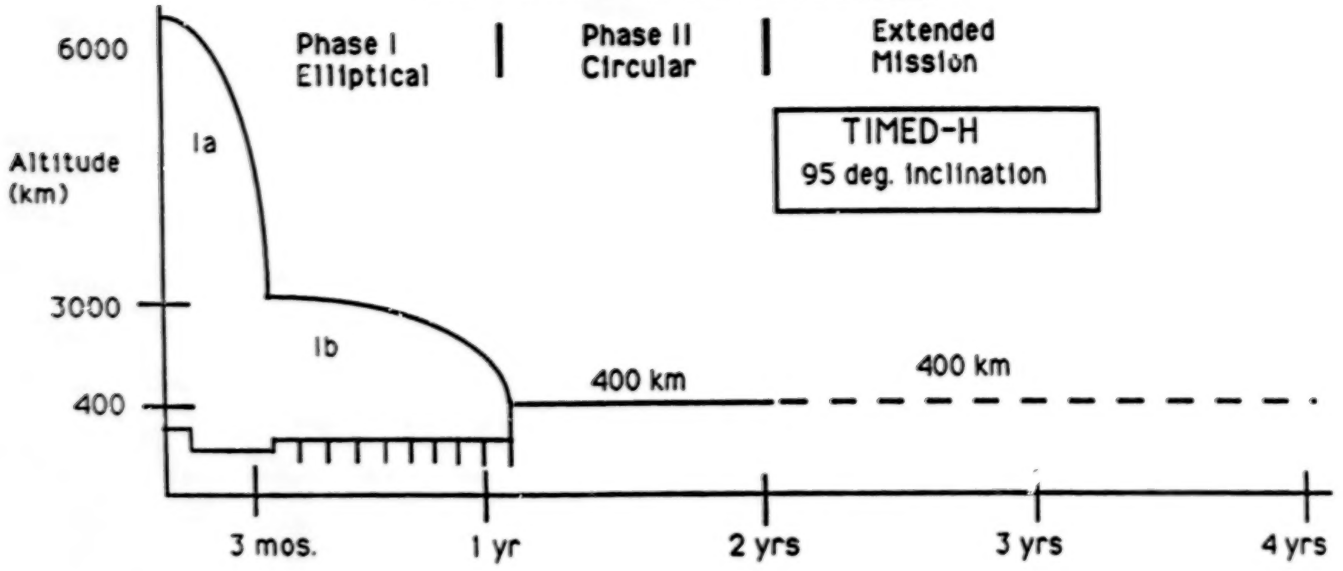


Figure 2-1

TIMED ORBITAL PROFILES



1a) One month checkout with perigee at 200 km. Then drop perigee to 132 km. Perigee moves from north pole to south pole.
1b) Perigee at 157 km, with monthly perigee dips to 132 km for 12 consecutive orbits.

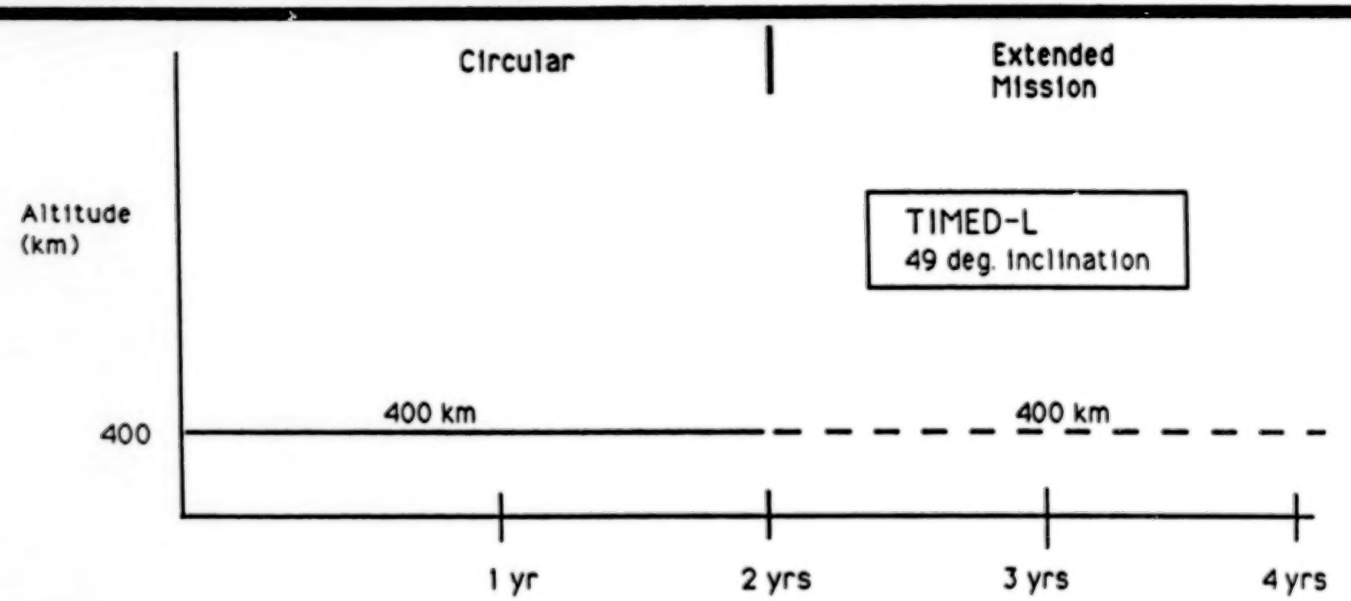


Figure 2-2

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ATMOSPHERIC DENSITY

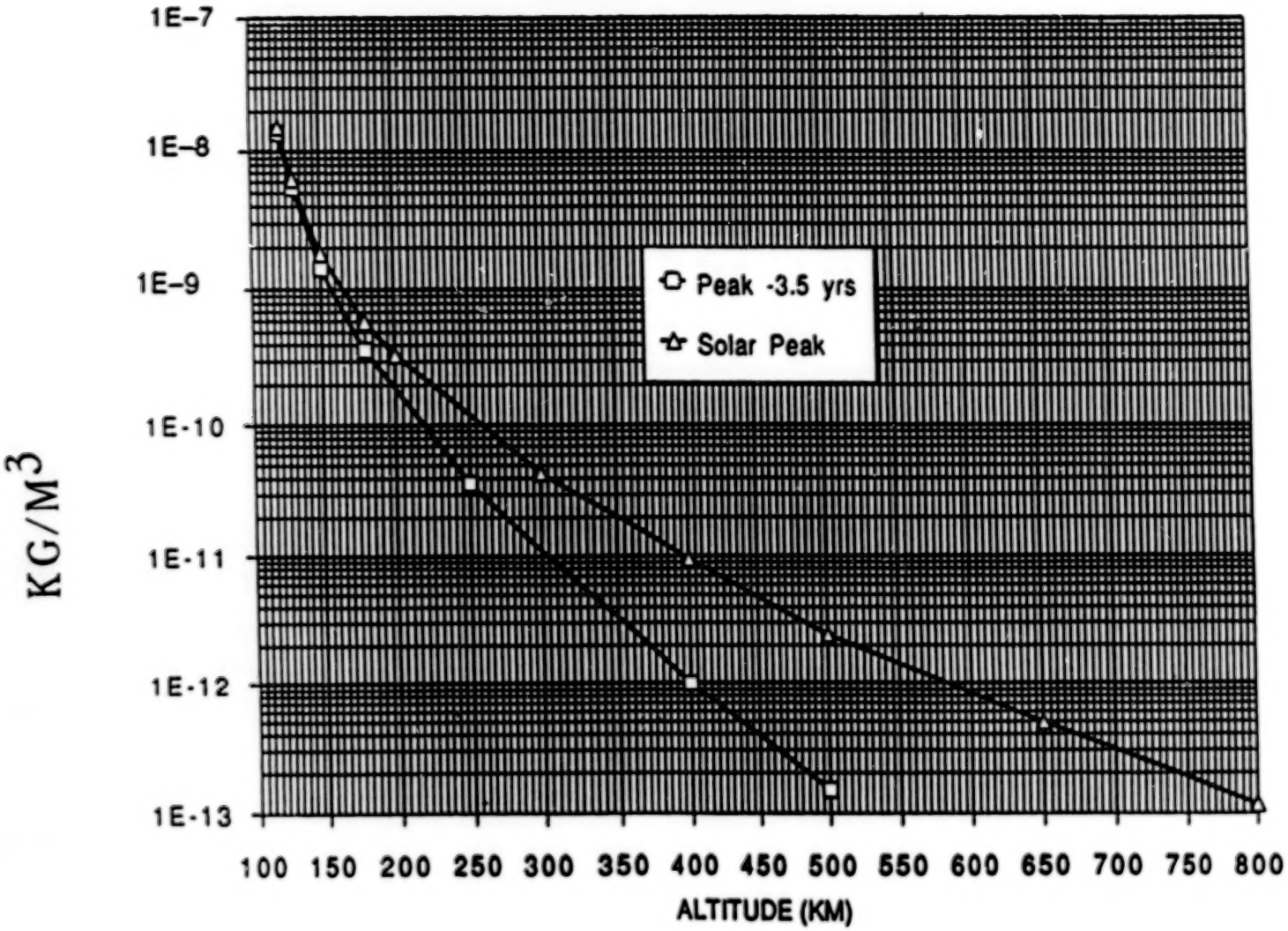

Figure 2-3

The solar beta-angle can take just about any value for either TIMED orbit. This means that there is nowhere to place a dedicated radiator that will not get sun on it at some time during the mission. This complicates the thermal design of the spacecraft, unless the spacecraft is turned every time the sun is in the orbit plane (beta = 0). A trade study examined the advantages and disadvantages of using a yaw turn maneuver to keep the sun on one side of the spacecraft. The study concluded that the yaw turn is the most cost effective approach. This always keeps the sun on one side of the spacecraft, allowing the other side to be dedicated to a thermal radiator. This approach has been employed successfully on such missions as the Upper Atmosphere Research Satellite (UARS), the Dynamics Explorer (DE) and the Earth Radiation Budget Experiment (ERBE).

The yaw turn keeps the thermal system simple, but has a negative effect on the ram facing instruments which will be turned around facing into the wake until the sun comes around to the orbit plane again. On TIMED-H, because of the slow precession of the orbit, the local time of launch can be picked so that almost the entire first year will pass before a yaw turn has to be made. This means almost the entire elliptical phase (the phase most interesting to the ram instruments) can be conducted with the ram instruments properly oriented.

The yaw turn is actually beneficial to the solar pointing instrument, since the sun will always be located on the same side of the observatory. The nadir looking instruments will notice a reversal in the direction of their scan, but they will be viewing the same swath of the Earth. The limb scanning instruments will be reversed so that the left limb now becomes the right limb, and vise versa, but this was not seen as a problem by the Science Definition Team.

2.2.1 Orbital Dynamics

The two spacecraft are placed into different orbits as shown in Figure 2-4. TIMED-H starts life in a high inclination, elliptical orbit giving it global coverage, including the high latitude regions. The latitudinal motion of perigee dominates over the local time precession rate of the orbit plane, so that perigee moves from pole to pole during the first six months of the mission with full periapsis coverage of all local times during the first year.

At injection, TIMED-H is given an apogee of sufficient height to satisfy the requirements of the auroral viewing instruments and with enough energy in the orbit so that perigee can be safely dropped to 130 km for three months with no need for propulsion. The energy in the orbit is used up like money in the bank, bringing the apogee down to around 3000 km after three months. To maintain apogee would require jet firings at perigee, where the in-situ instruments are taking their data.

Once TIMED-H apogee reaches 3000 km, the plan is to raise perigee to around 150 km. For the remainder of the first year, TIMED-H will only conduct monthly excursions down to 130 km for 12 consecutive orbits. This will slow the apogee decay and extend the elliptical phase for one year, but some apogee maintenance will be necessary.

At the end of one year in the elliptical phase, the TIMED-H orbit will be circularized at an altitude of 400 km. It will be maintained there for the duration of the mission.

TIMED-L is placed into a lower-inclination, circular orbit at an altitude of 400 km where it remains for the entire mission. The requirement on the inclination is that it be low enough to yield a full diurnal cycle in less than two months and high enough to allow limb scanning of the auroral oval.

INITIAL TIMED ORBITS

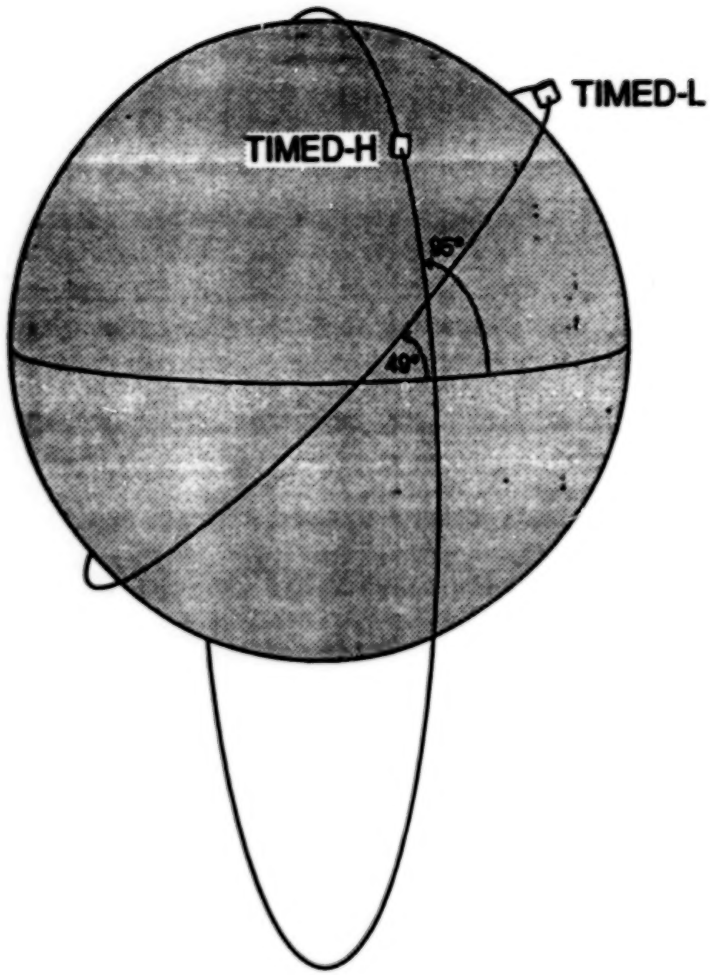

Figure 2-4

2.2.2 Control Concept

The control concept for TIMED is a nadir pointing control system (see Figure 2-5). This means that the observatory maintains an average rotation rate of one revolution per orbit about the orbit normal while it maintains one side pointing along the local vertical. In the elliptical orbit on TIMED-H, the velocity vector will move in body coordinates as the observatory follows nadir. This is because the velocity vector and the local vertical do not remain normal in an elliptical orbit. In fact, they are only normal to each other at apogee and at perigee.

The in-situ instruments want to look into the velocity vector, whereas the remote instruments want to view the limb of the Earth. It is impossible to satisfy both requirements simultaneously. Fortunately, the in-situ instruments are interested in the region around perigee. Here, the velocity vector is within a degree of the normal to the ram face of the observatory for approximately 5.0 degrees of true anomaly on either side of perigee for the worst case orbit eccentricity of 0.311 for a 6000 km by 130 km orbit. If more accuracy is required, it would be possible to have a velocity following algorithm using the active pitch loop. The remote instruments are generally not interested in data around perigee, so switching control from nadir to velocity pointing around perigee would not upset their data taking.

A bias momentum control system was determined to be the most cost effective attitude control system for TIMED. Redundant momentum wheels running at approximately half speed provide stability in the roll and yaw axes, and active control in pitch. Magnetic torque rods unload momentum, provide attitude error correction and perform the yaw turn maneuver. The propulsion system is used for orbit adjust and as a backup for attitude maneuvers and for attitude control.

Two Earth scanners provide pitch and roll attitude with respect to a nadir reference system to an accuracy of 0.3 degrees (3 sigma). Two digital sun sensor heads provide yaw attitude knowledge to the same accuracy. This is adequate for most of the TIMED instruments which require attitude knowledge information to an accuracy of 0.1 degrees (1 sigma), or 0.3 degrees (3 sigma). However, a few strawman instruments indicated a need for attitude knowledge accuracy of 0.05 degrees. This requirement would have driven the design away from Earth scanners toward a more expensive system based on one or more star trackers and perhaps an inertial reference system (gyros). It was felt that such a system was not in keeping with the philosophy of an intermediate class mission and, if only a few instruments needed the tighter knowledge, perhaps they should supply their own attitude sensor. This would be more cost effective than modifying the entire observatory. Words to this effect were put in the Announcement of Opportunity: "Proposers that require...pointing knowledge better than 0.3 degrees should include their requirements (along with justifications) in the proposal and address how these requirements could be accommodated within their proposed instrument and/or a spacecraft provided system." [Reference 4, p.9] It was decided that we would stick to the Earth scanner approach for Phase A and re-examine the question in Phase B when the final instruments had been selected.

2.3 Instruments

Table 4-3 shows the strawman set of instruments for the Phase A design. Both remote sensing instruments and in-situ instruments will be flown on each spacecraft. TIMED-H carries 9 instruments and TIMED-L carries 6. All of the TIMED-L instruments are copies of instruments on TIMED-H except for the Infrared Limb Sounder, which only flies on

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TIMED BIAS MOMENTUM CONTROL

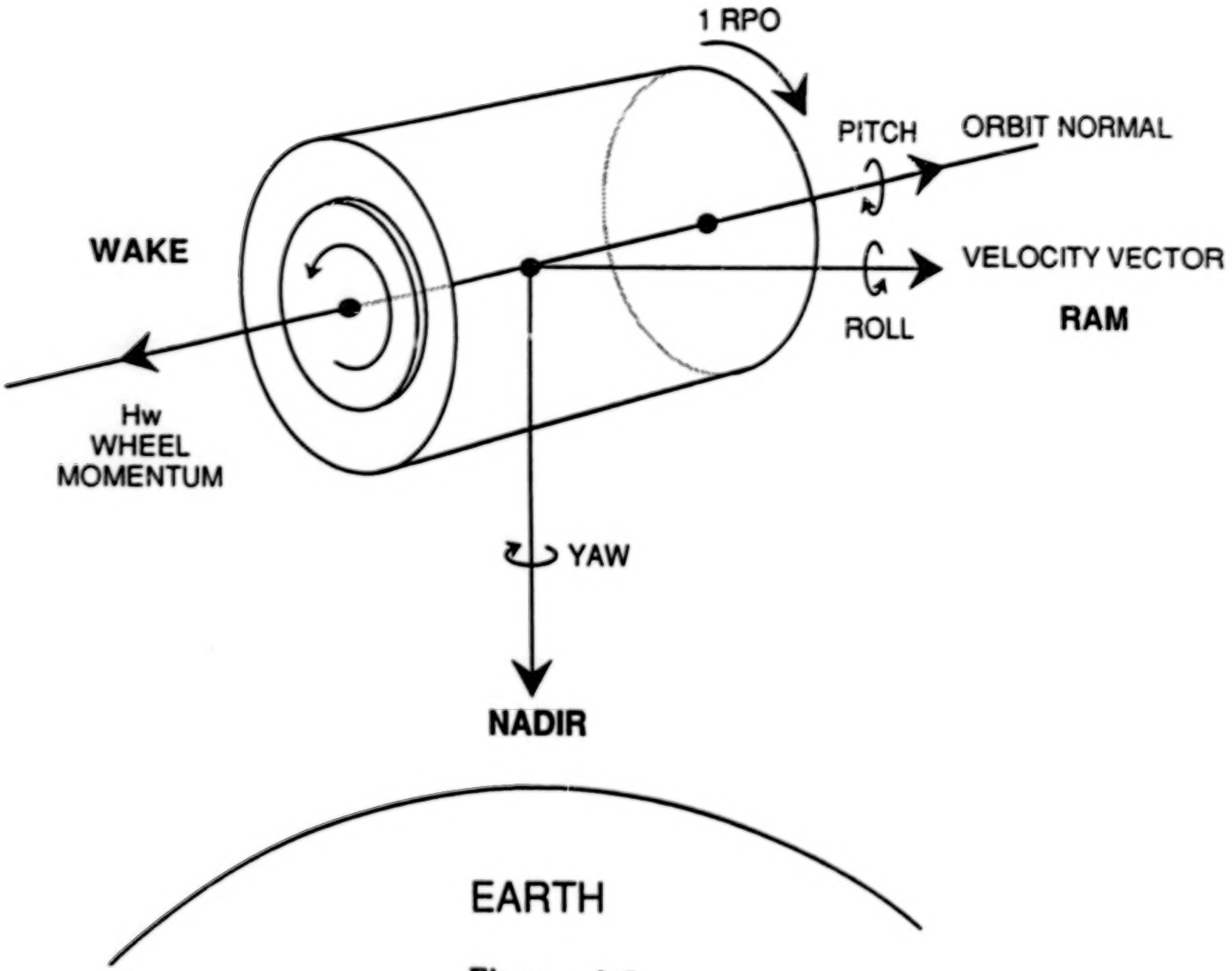

Figure 2-5

Extreme Ultraviolet (SEUV) requires a two-axis gimbal pointing platform so that it can point to the sun independent of the observatory orientation. The design of this pointing platform was never worked during Phase A, although its weight estimate and cost were carried in the total for the instruments.

The TIMED strawman instruments are largely based on heritage instruments and, except for the IRLS, have relatively low power, weight and data rate requirements. The final selected option has no boom instruments. These instruments were considered on more costly options, but were dropped when the Science Definition Team descoped the science. The deletion of the boom instruments was beneficial to the spacecraft design.

2.4 Spacecraft Summary

The two TIMED observatories are identical in design, except for small differences in the accommodation of the instruments. The biggest difference is that TIMED-L has a small deployable array to boost the power output because of the IRLS instrument (see Figure 2-1). This meant that the higher power requirement for the IRLS on TIMED-L would drive the power requirement for both spacecraft.

The spacecraft are 2.1 meters in diameter and 1.7 meters long with nine sides. Fully fueled, TIMED-H weighs 946 kg and TIMED-L weighs 883 kg. The power requirement on TIMED-H is 215 watts and on TIMED-L it is 283 watts. The higher power requirement on TIMED-L is due to the IRLS instrument with its mechanical coolers. In order to keep this imbalance from driving the size of the common spacecraft design, a small deployable solar array was added to TIMED-L to boost its power. This allowed both spacecraft to shrink in size since all other solar arrays are body mounted. Because TIMED-L is in a circular orbit at 400 km, and because the array is essentially edge-on to the velocity vector, there is a minimum impact on the attitude control system with the deployed array.

The observatory interior is based on a modular design with a lower deck for mounting spacecraft components and for the cold-side radiator, an upper deck for mounting the instruments and a central support truss which supports the propulsion module with its six tanks.

2.5 Integration and Test

The two TIMED spacecraft will be integrated in parallel, with one of the spacecraft preceeding the other by three months. Although there has to be two integration teams, there are some efficiencies realized from this approach. For example, only one subsystem lead engineer is needed in each discipline since one engineer can maintain cognizance of both integration flows. Also, some hardware can be shared between the two spacecraft integrations. It was determined that only one shipping container is required to ship the two spacecraft to the launch site, since they are shipped at different times. Also, if an instrument appears on both spacecraft, only one mass mockup is needed since the spacecraft structural tests are run at different times.

The parallel integrations also give the Project some flexibility when problems arise. Normally, the priority will be given to the first vehicle by installing and testing components and subsystems there first. But, if a problem occurs, it may be necessary to divert hardware scheduled for the second vehicle and use it on the primary vehicle to maintain schedule. The modularity of the design permits some additional parallel integration in the early stages of integration and test since the spacecraft and instrument modules can be worked separately.

2.6 Science Data System

The science data system is responsible for the processing, storing, archiving and dissemination of instrument data. It is part of the overall TIMED ground system. The other part of the ground system is the institutional ground system, which is responsible for mission operations including health and safety of the spacecraft and instruments and science data acquisition.

A comprehensive trade study was conducted to establish the architecture for the science data system. The selected architecture is centered around a facility called the TIMED Science Data Room (TSDR) which will be located at GSFC and performs the functions of science planning, science operations scheduling, instrument monitoring, science product generation through Level 3, science data storage and distribution, and data and information management. The TSDR configuration is workstation-based with one product generation workstation allocated to each instrument.

The other side of the science data system is located at the principal investigators' (PI's) facilities. The principal investigators will be responsible for algorithm development, science planning, instrument monitoring, science operations, science product generation and Level 4 processing.

The PI's facilities are connected to the TSDR through an established science data network, such as the Space Physics Division NASA Space Internet (NSI) network.

2.7 Mission Operations

The other function of the TIMED ground system is to support mission operations. The objective of mission operations is to support science activities and maximize science data acquisition while caring for the health and safety of both observatories, including analysis of flight dynamics for orbit determination and orbit adjust maneuvers.

The groundrule used in the Phase A Study was to make maximum use of the GSFC institutional resources for mission operations in order to reduce risk and contain cost.

The Mission Operations Control Center (MOCC) will be located at GSFC and will serve as the central point for all satellite control and monitoring. The MOCC will be responsible for coordination between the mission operations and the science operations which are conducted in the TSDR. It will handle all command and control functions, including command validation, constraint checking and load generation. The MOCC will also monitor spacecraft and instrument health and safety, perform mission planning and scheduling, and do science data recovery.

The Flight Dynamics Facility (FDF) will provide all orbit-related data and orbit adjustment procedures. It will also support the attitude determination function through attitude sensor data processing, including sensor calibration and alignment.

The Data Capture Facility (DCF) will receive the recorded science data and perform Level 0 processing before it is sent to the TSDR.

The Command Management Services (CMS) may reside in the MOCC or the Command Management Facility (CMF). It will provide command memory management, constraint checking, and command sequence generation. It will also aid in the development of operation plans.

The TIMED architecture uses a single ground station, the Wallops Orbital Tracking Station (WOTS), for all science data acquisition. WOTS and the Tracking and Data Relay Satellite System (TDRSS) will both be used for command uplink at 2 Kbps and for engineering data downlink at 1 Kbps.

The maximum daily data accumulation is 2.36 Gbits on TIMED-H and 1.4 Gbits on TIMED-L. This is with the instruments operating at 100 percent duty cycle with a 25 percent margin added. To dump this data to the ground requires 26.3 minutes/day on TIMED-H and 15.6 minutes/day on TIMED-L at the 1.5 Mbps rate. Analysis of the contact times shows that coverage at WOTS is sometimes marginal to retrieve all the data at the 1.5 Mbps rate when the instruments are running at 100 percent duty cycle. The minimum daily contact time on TIMED-H drops to 22.9 minutes when TIMED-H is in a circular orbit at 400 km. In these cases, the downlink rate can be switched to 3.0 Mbps. WOTS can receive at this rate and the only drawback is that the transmission to GSFC can only support the 1.5 Mbps, so data will not be visible in real time.

In addition to WOTS, the JPL's Deep Space Network (DSN) will be used as a backup in the event there are contact problems or scheduling problems with WOTS. However, there is a slower transmission rate between the DSN stations and GSFC of only 224 Kbps.

2.8 Launch Vehicles

To satisfy the two orbital inclinations for the TIMED Mission, requires an East Coast and a West Coast launch. The selected launch vehicle is a Delta II for both launches. This choice was based on the fact that the Delta II is the only launch vehicle that meets the TIMED requirements and also meets the NASA guidelines for the selection of a medium expendible launch vehicles (MELV).

The Delta 7320, with 3 solid rocket motors, will meet the requirements for both launches. The West Coast launch of TIMED-H into a near-polar orbit, requires the greatest boost. The Delta 7320 can place 1,118 kg into the required orbit. With the TIMED-H total wet mass of 946 kg, the lift margin is 18 percent. Because of the low altitude and low inclination, the TIMED-L orbit has considerably more margin. The Delta 7320 can lift 2,491 kg into the TIMED-L circular orbit. The total wet mass of TIMED-L is only 883 kg.

The TIMED observatories fit within the Delta 9.5 foot (2.9 m) diameter fairing. A 6019 Payload Attach Fitting (PAF) is the mechanical interface between each TIMED observatory and the booster. It provides a three-point mount which transfers the launch loads directly into the TIMED triangular structure.

3.0 SCIENTIFIC OBJECTIVES

3.1 Introduction

The TIMED mission is designed to investigate the mesosphere and lower-thermosphere/ionosphere (MLTI), extending from about 60 to 180 km, which is considered the "least explored region in the Earth's atmosphere" [Reference 1]. Two spacecraft in low and high inclination orbits would carry instruments for in-situ measurements and remote sensing.

Programmatically, TIMED will bridge the gap between the Upper Atmosphere Research Satellite (UARS) which is studying the dynamics, energetics and chemistry of the stratosphere and International Solar Terrestrial Physics (ISTP) which will in part study the magnetosphere and atmosphere. The TIMED instrumentation heritage for in-situ measurements is primarily in the Atmospheric Explorer (AE) and the Dynamics Explorer (DE) explorers, and for remote sensing in the Solar Mesospheric Explorer (SME) and UARS.

3.2 Science Background

The MLTI region to be investigated by TIMED is too high for balloon and aircraft experiments and generally too low for in-situ measurements from high flying spacecraft without on-board propulsion (see Figure 3-1, taken from the SDT Report [Reference 2]). Thus it has been explored far less than the more accessible adjacent regions and in particular that of the upper thermosphere and ionosphere which was studied extensively. While the TIMED MLTI region is seen as one of the last frontiers in our exploration of the Earth's atmosphere, it is extremely complex and challenging scientifically; and it is also of considerable societal interest because of the anthropogenic and solar terrestrial effects that are known to occur.

Scientifically, the MLTI region is characterized as one of transitions, in which many of the important processes are known to change fundamentally; and what causes these changes is not fully understood, nor is it known where they occur. For example, in the mesosphere, turbulent mixing or eddy diffusion dominates such that the relative concentrations of atmospheric species tend to be independent of altitude. Above what is referred to as the turbo-pause, molecular diffusion begins to dominate so that the composition changes drastically with altitude, the heavier ones concentrated at the bottom and the lighter ones at the top. The nature of turbulent mixing is not understood and it is not known where the turbo-pause transition to molecular diffusion does occur or how it changes with time; e.g., annually.

Associated with eddy and molecular diffusion, are processes of heat and momentum transfer which impact the energetics and dynamics of the region. Thus, radiative processes tend to dominate over eddy heat conduction in the mesosphere, while molecular heat conduction becomes increasingly more important at higher altitudes such that it dominates in the upper-thermosphere. Dynamically, the consequence is that in the mesosphere the winds tend to be geostrophic due to the Coriolis force from the Earth's rotation which causes them to blow in a direction across pressure gradients. While in the thermosphere, the winds become increasingly more ageostrophic, directed along pressure gradients, due to the importance of molecular viscosity.

The energy sources from the sun that drive these processes also change throughout the region. In the mesosphere and stratosphere, the absorbed radiation is in the UV range

PRIMARY ALTITUDE REGIMES FOK TIMED SCIENCE INVESTIGATION

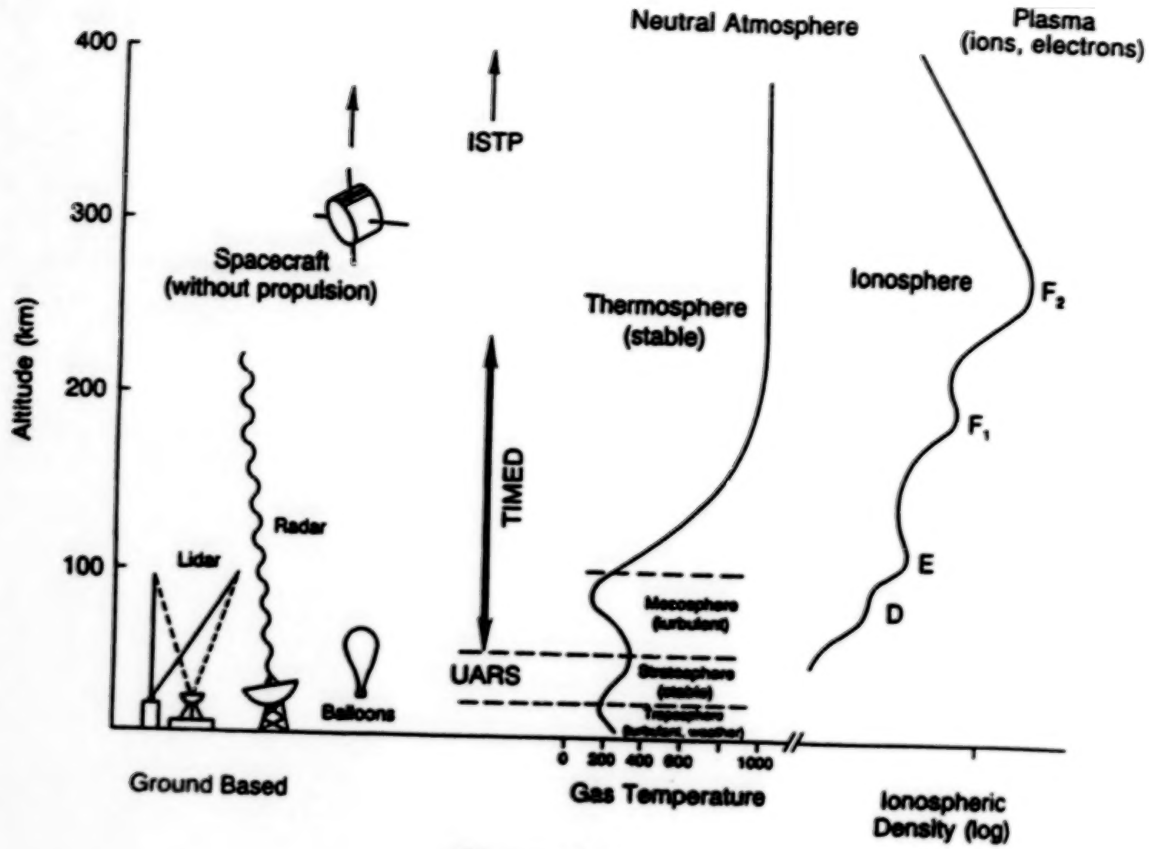

Figure 3-1

which varies only on the order of 10 percent with solar activity. In the tenuous thermosphere at higher altitudes, the solar input derives primarily from the EUV radiation and that can vary by more than a factor of two over a solar cycle. This variable EUV flux does not carry much of the solar energy; but, its short wavelength photons are relatively energetic. Thus, the specific heat input per gas molecule is large and so are its variations with solar activity, which accounts to a large extent for the high temperature and its variability that characterize the thermosphere. The energetic photons in the EUV also ionize the atmosphere to produce a plasma of ions and electrons that forms the ionosphere. Although the density population of this ionosphere is comparatively small, its effects on the parent neutral atmosphere are large due to the presence of the geomagnetic field and the electrodynamic solar wind interaction with the magnetosphere.

3.3 TIMED Science

Paraphrasing the SDT Report [Reference 2], the scientific objectives for the TIMED mission are stated as follows:

- To understand the physical and chemical processes in the coupled MLTI region (between about 60 and 180 km altitude)

- To understand the coupling between chemistry, radiation and dynamics (tides, planetary waves, gravity waves, unorganized motions and large scale circulation), and the resultant effects on the energetics of the MLTI

- To understand the odd oxygen, odd nitrogen and odd hydrogen chemistry affecting the energetics of the MLTI

- To understand the role of the lower-ionosphere in the global electric circuit

- To investigate the anthropogenic effects in the MLTI and establish a baseline for future investigations.

Some of the specific phenomena and processes the TIMED mission will investigate are illustrated in Figure 3-2; which also depicts the two spacecraft in high and low inclination orbits with instrument capabilities for in-situ and remote measurements. They show, specifically, the average temperature distribution in the atmosphere above 50 km, covering the globe from the polar winter hemisphere on the left to the polar summer hemisphere on the right. The importance here is that below 100 km, the lowest temperatures in the Earth's atmosphere are observed and they occur in summer and not in winter, contrary to expectations from radiative considerations which hold in the adjacent regions below and above. This temperature anomaly is not fully understood. A number of processes have been invoked, such as large scale energy transport from the summer to the winter hemisphere, with adiabatic cooling and heating in these regions respectively, and thermospheric transport of atomic oxygen which holds chemical energy that is released in winter.

PHENOMENA AND PROCESSES INVESTIGATED BY TIMED

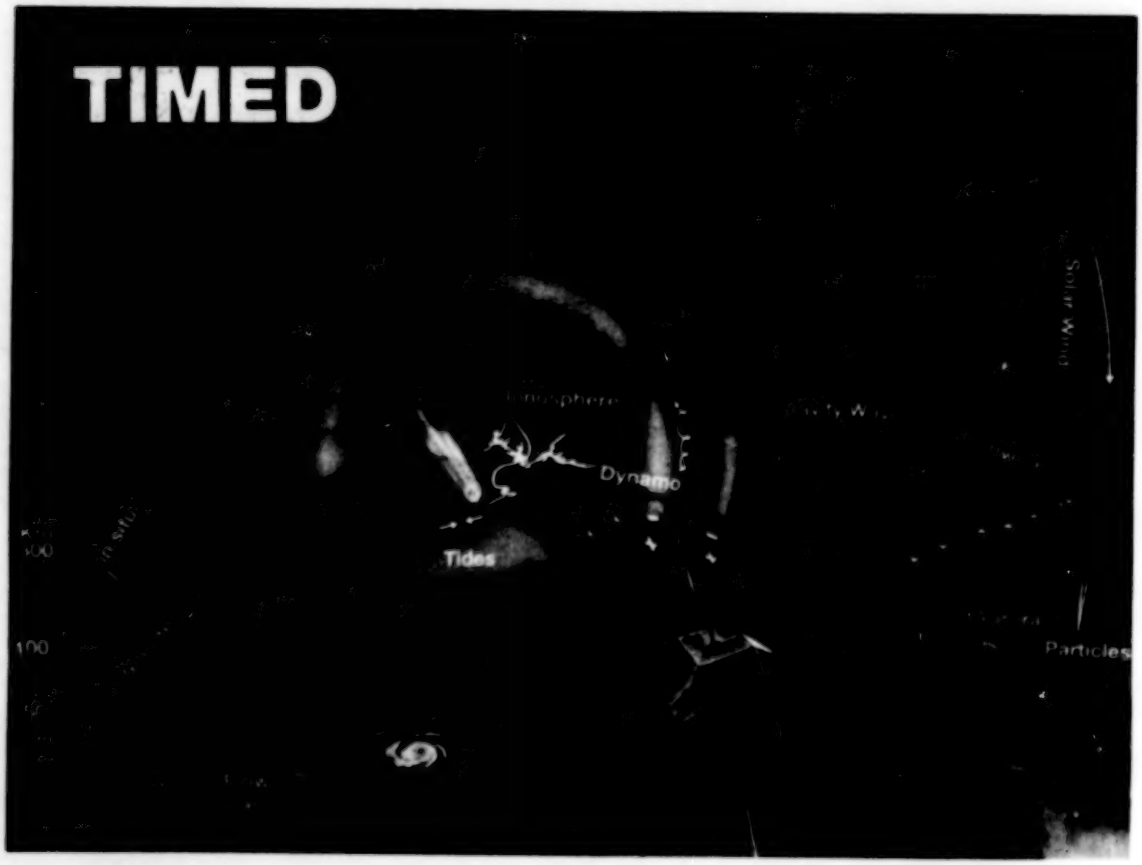

Figure 3-2

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Associated with this anomalously low temperature in the summer hemisphere of the upper mesosphere are noctilucent or polar mesospheric clouds which are a mystery. These clouds have not been observed before the turn of the century and have doubled in occurrence during the last 20 years. One of the explanations involves methane produced in the biosphere which is transported up and may be oxidized to form water, a process which may also be involved in the destruction of ozone. Related to that is another anthropogenic effect, related to increased abundance of carbon dioxide (CO_2). CO_2, a greenhouse gas, is believed to raise the temperature in the troposphere near the ground where it effectively traps the outgoing infrared radiation. At higher altitudes, however, CO_2 becomes an effective radiator, thus lowering the temperature which has been seen to decrease by 2-3 degrees K over the last decade. Such a long term decrease in the mesospheric temperatures could contribute to the formation and equator-ward spread of noctilucent clouds which may eventually affect the climate here on Earth. The important time constants of these processes are relatively short at higher altitudes. Thus, the anthropogenic effects in the mesosphere may herald changes that occur later near the ground. While the limited lifetime of the TIMED mission will not allow us to monitor these changes if they occur, the measurements will be important for developing an understanding of the basic processes involved and to provide a baseline for future investigations.

Figure 3-2 also illustrates the rich phenomenology related to the dynamics of the MLTI region. The background temperature distribution already was discussed. Shown on the left is a hurricane representative of a weather system in the troposphere which will spawn gravity waves. These waves propagate up and encounter an atmospheric density that exponentially decreases with altitude. Like in ocean waves rushing towards the shallow waters near the shore, the amplitude increases with altitude to conserve energy, and eventually the waves break up into turbulence and are dissipated. As discussed earlier, turbulent mixing and the transition to molecular diffusion that characterizes the turbo-pause near 100 km is of first order importance for the structure of the thermosphere above. The waves thus described also carry momentum, in particular the shorter wavelengths, and accelerate the large scale meridional (north-south component) winds, which in turn can redistribute energy to produce the anomalously low temperatures in the summer hemisphere as discussed earlier. The propagation characteristics of gravity waves are affected by the large scale zonal circulation in the east-west direction which changes direction seasonally over the course of a year and acts like a filter causing modulations in the wave amplitudes. Such filtering has been invoked to explain seasonal variations with north-south asymmetries observed in the density and temperature of the thermosphere. Strong gravity wave activity below the turbo-pause has been observed with ground based lidar measurements.

In the middle of Figure 3-2, the phenomenon of atmospheric diurnal tides is illustrated which is of great importance to the TIMED mission. Diurnal tides are driven by the variable solar heating over the course of a day, which peaks in the early afternoon and is absent at night. Like gravity waves, these tides can propagate into the atmospheric regions at higher altitudes and they also grow in amplitude up to about 100 km where dissipation becomes important. Tides are excited by the absorption of the visible radiation in the water vapor layer near the ground and by the UV radiation absorbed in the ozone layer around 50 km. They are characterized by sub-harmonics of the diurnal (24-hour period), semi-diurnal (12-hour), terdiurnal (8-hour), etc., components which differ markedly in their latitude dependencies. The ability of these tides to propagate upwards depends on a number of factors, such as their vertical wavelengths which are relatively large for the semi-diurnal tide, with the result that this component tends to prevail at higher altitudes (as illustrated with the two maxima in Figure 3-2). Tides generated in the thermosphere by the absorbed EUV radiation cannot propagate because of dissipation due to molecular viscosity; thus, the

diurnal tidal component tends to prevail there. Ground based observations with radars show that the tides in the 80 to 110 km region greatly vary with season and, surprisingly, over time scales on the order of weeks which is not understood. Even at thermospheric altitudes, there is evidence of this variability as seen in the data from AE-E.

The diurnal tides are known to be important for the electrodynamics of the ionosphere. In the presence of the geomagnetic field and ionospheric conductivity, the diurnally varying wind field generates dynamo electric fields that drive currents which are revealed in magnetic signatures. This dynamo electric field causes the ionospheric plasma to drift across the geomagnetic field and that in turn changes the ionospheric density distribution, triggering instabilities and the formation of plasma plumes that are observed in the equatorial region. The drifting plasma also accelerates the neutral atmosphere through collisions, thus providing an electrodynamic feedback for the wind field.

The last topic to be addressed here concerns the solar wind interaction with the Earth's magnetosphere which induces a large scale electric field that virtually dominates the electrodynamics and energetics of the thermosphere and ionosphere at high latitudes. This electric field causes the ionospheric plasma (electrons and ions) to drift across the Earth's magnetic field attaining large flow velocities. Thus, a convection pattern with a double cell vortex circulation is set up, which is illustrated in the right hand corner of Figure 3-2. The drifting ions then accelerate the neutral atmosphere and produce large wind velocities, often exceeding 1000 m/s, which is at least a factor of 10 larger than those observed in other regions of the Earth's atmosphere. At altitudes around 130 km where the ionospheric plasma and the neutral atmosphere are tightly coupled through collisions, the electric currents driven by the solar wind-induced electric fields are dissipated, and magnetic disturbances are prominent manifestations. These currents provide an important energy source for the neutral atmosphere through Joule heating, a process which dominates the energetics of the polar region and can affect the thermosphere globally through large scale transport processes. Precipitating particles associated with auroral emissions ionize the atmosphere and increase its electrical conductivity, which tends to enhance the current flow and, with it, the Joule heating.

Given the transient nature of the solar wind interaction with the magnetosphere and the patchiness of the precipitating auroral particles, the polar or auroral regions of the thermosphere and ionosphere are changing continually and often dramatically. These conditions are conducive to the generation of gravity waves which have been observed abundantly. While these processes have been the subject of intense studies carried out with DE, they are by no means fully understood. The bulk of magnetospheric energy is deposited in the lower thermosphere, and this region has not been covered with satellite observations. The TIMED Mission with its complement of instruments for in-situ measurements and remote sensing is specifically designed to provide the necessary observations. It will not only cover the critically important altitude regime of the lower thermosphere, but will also provide large scale, or even global scale, coverage which is essential for studies of the atmosphere dynamics.

3.4 Baseline Mission Scenario

The science review on the previous pages underscores the need for two spacecraft, one in a high inclination orbit that covers the high latitude region and provides global coverage, and another one in a mid to low inclination orbit with a high enough precession rate that facilitates good coverage of the local time variations to study tidal phenomena. In the baseline mission scenario, the SDT thus chose inclinations of 95 degrees for TIMED-H and 49 degrees for TIMED-L. The requirement of performing in-situ measurements in the lower thermosphere stipulated the need for on-board propulsion which in turn provided the

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flexibility to design a mix of phased orbital scenarios, reproduced in Figure 3-3, that would be ideally suited to accomplish the mission objectives. After a careful review of the science driven measurement requirements and measurement techniques, the SDT developed strawman instrument payloads for in-situ measurements and remote sensing which are reproduced in Table 3-1. The chosen instruments had spaceflight heritage.

BASELINE MISSION SCENARIO

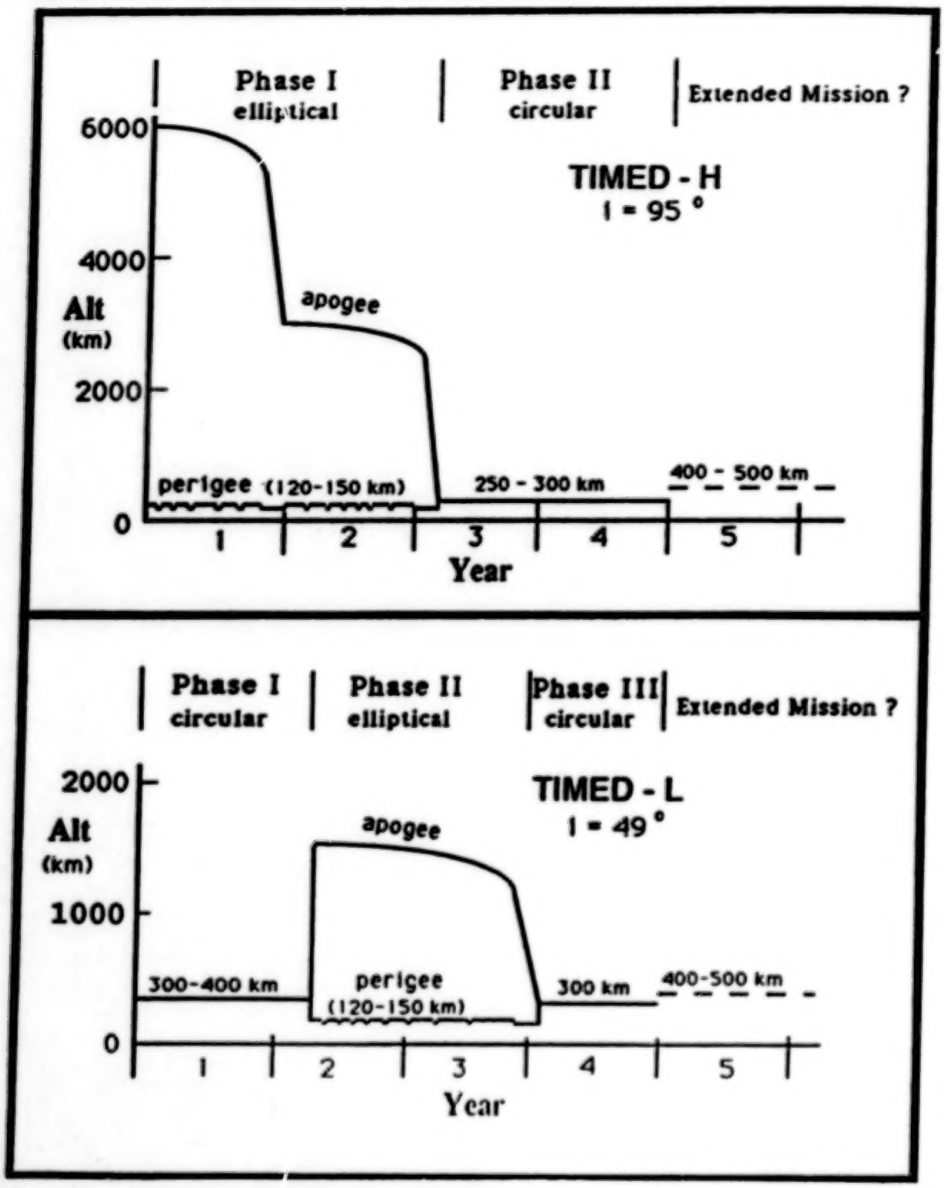

Figure 3-3

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Primary Strawman Instruments for TIMED-H and TIMED-L

Instruments	Acronyms	Measured Parameters	TIMED-H	TIMED-L
A. Remote Sensing				
Fabry-Perot Interferometer	FPI	Wind, temperature and emission rate profiles	X	X
Imaging Photometer	IP	Wave structures	X	X
Global UV Airglow Imager	GUVI	Global imaging of atmospheric UV emissions	X	-
Infrared Limb Sounder	IRLS	Temperatures (from CO_2 emissions); 0_3, OH and NO emissions	-	X
Near Infrared Spectrometer/Photometer	NIRSP	O_2 ($^1\Delta$), 0_3, H_2O, OH vibrational	X	-
Solar EUV Spectrometer	SEUVS	Spectral solar EUV and x-ray fluxes	X	-
UV Spectrometer	UVS	O_3, NO_x, other minor constituents, temperature	X	X
B. In-Situ				
Neutral Mass Spectrometer/Wind and Temperature	NMSWT	Neutral composition, winds and temperature	X	X
Ion Drift Meter/Retarding Potential Analyzer	IDM/RPA	Ion drift/electric fields, ion temperature and composition	X	X
Langmuir Probe	LP	Total electron/ion density and temperature	X	X
Energetic Particles Analyzer	EPA	Auroral electron and ion energy fluxes (10eV-30keV)	X	-
Magnetometer[2]	MAG	Currents	X	-
Ion Mass Spectrometer	IMS	Ion composition	X	X
Electric Field Detector/Plasma Wave Experiment[2]	EFD/PWE	3-axis AC and DC electric fields	X	
Accelerometer	ACC	Total neutral density	X	X

1 Not needed on TIMED if done simultaneously on other mission(s)
2 Contingent on feasibility of booms at low periapsis altitude

Table 3-1

The requirements for the baseline mission are summarized as follows:

- Global coverage, particularly the high latitude regions, which requires a high inclination orbit (95 degrees)
- Rapid local time precession of the orbit to map tidal structures, which requires a low inclination orbit (49 degrees)
- Remote sensing down to 60 km, in-situ measurements into the lower thermosphere (130 km)
- Propulsion capability for phased orbital adjustments
- Combination of elliptical and circular orbits for remote sensing and in-situ measurements
- Three-axis stabilization for accurate remote sensing
- Mission lifetime of 4 years

The Pre-Phase A Study conducted with this mission scenario assumed that both spacecraft would be essentially identical in order to contain cost and that an aerodynamically clean design with body mounted solar cells would be required to reduce atmospheric drag. The latter eventually stipulated that power to the instrument would have to be limited to a 25 percent duty cycle at the end of mission life.

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A preliminary cost analysis of this baseline mission indicated that it could not be implemented under the cost guidelines set forth by the Space Physics Division, and the SDT was, therefore, asked to focus the science objectives and develop some alternative lower cost mission scenarios.

3.5 TIMED Alternative Mission Plan

Even though the SDT had exercised a great deal of discipline, the above discussed scientific objectives and the baseline mission scenario were broad in scope and, understandably, there was a great deal of resistance to pare them down. Considering how strong the coupling is between the regions of the atmosphere and between its neutral and ionized components, and how interwoven the different science areas of energetics, dynamics, chemistry and electrodynamics are (illustrated in Figure 3-4), it was difficult to develop a science focus that preserved much of the integrity originally intended for the mission. Different routes were taken to attack the problem from different directions. One was to view the atmospheric regions of interest in programmatic terms, so that a prioritization could be developed which is shown in Figure 3-5. Here, the central argument was that no other mission before had ever addressed the important coupling between the mesosphere and thermosphere, justifying this to be of highest priority. Related to that is also the coupling between the lower thermosphere and the lower-ionosphere, which would be next in priority. Although SME had given us a glimpse of the mesosphere and the UARS mission would also shed some light on it, this region of the atmosphere is far less understood than the thermosphere and ionosphere above which have been studied extensively with in-situ measurements, although mostly at higher altitudes.

The prioritization of the science objectives was finally carried out in terms of the science areas the SDT had originally adopted in its report [Reference 2]: energetics, dynamics, chemistry, and ionospheric structure and electrodynamics. Choosing from a series of 25 key scientific questions that covered these areas, the consensus of the team was expressed in a list of prioritized science objectives which were published in the Announcement of Opportunity [Reference 4] and are presented in Table 3-2.

The highest priority science objective and the prioritized science areas are summarized in Tables 3-3 and 3-4, respectively.

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CONTRIBUTING SCIENCE AREAS

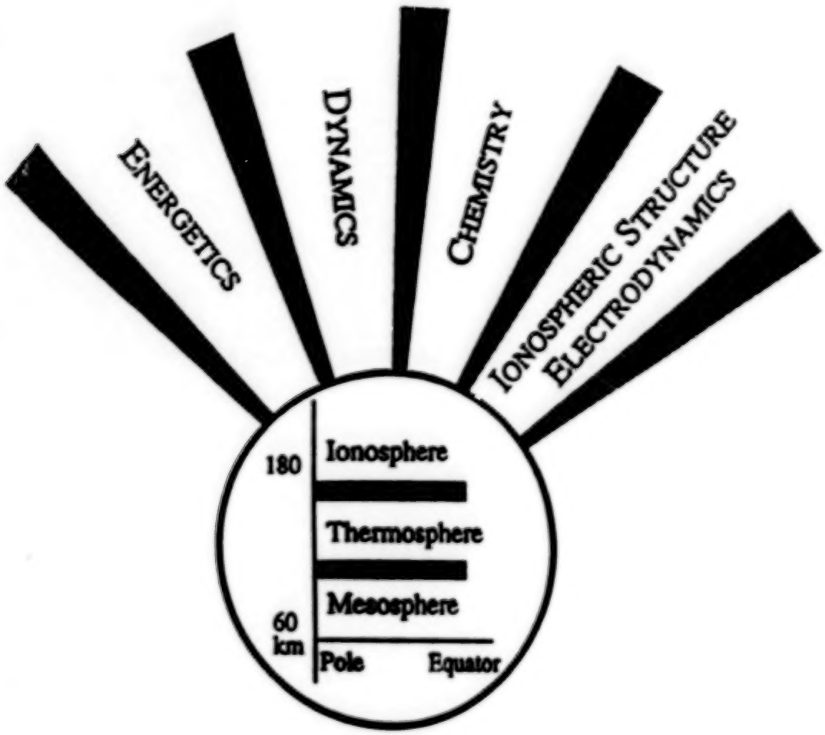

Figure 3-4

ATMOSPHERIC REGIONS OF INTEREST

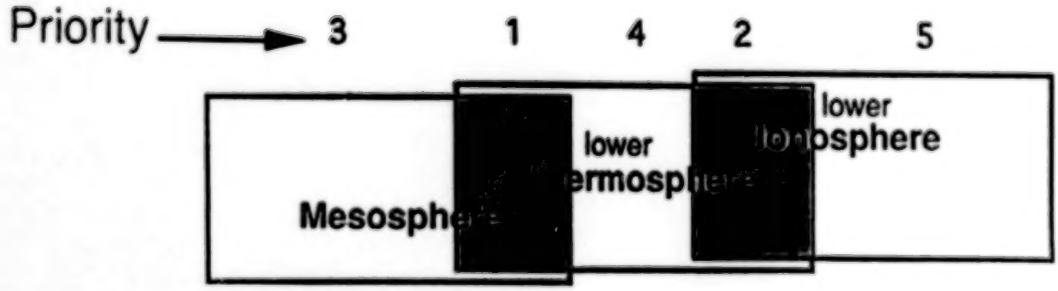

Figure 3-5

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Ranked Science Objectives for Alternative Mission Scenarios

Priority Ranking	Science Area	TIMED Science Objectives
1	energetics	Determine the temperature, density, and wind structure of the MLTI, including the seasonal and latitudinal variations
2	energetics	Determine the relative importance of the various radiative, chemical, electrodynamical, and dynamical sources and sinks of energy for the thermal structure of the MLTI
3	dynamics	Determine the sources and magnitudes of gravity waves and planetary waves (extra-tropical and equatorial) in the MLTI region, their spatial and temporal variability, and their forcing of the large-scale circulation, and the thermal and constituent structures
4	dynamics	Determine the mean global tidal structures, their annual and height variations, and their effects on the large-scale circulation in the MLTI
5	dynamics	Determine the relative importance of the interactions occurring among the different wave motions (tidal, gravity, and planetary) and the extent, causes, and consequences of their variability
6	chemistry	Determine the spatial and temporal variability of the radiatively significant minor species and the role that atomic oxygen plays in influencing the radiative balance through collision energy transfer
7	ionosphere	Determine the relative importance of the disturbance dynamo, tidal dynamos, and magnetospheric penetration fields in determining the global electrodynamics at mid and equatorial latitudes
8	chemistry	Determine the spatial and temporal variability of the odd-oxygen, odd-hydrogen, and odd-nitrogen compounds in the MLTI
9	ionosphere	Determine the roles of neutral winds and electric fields in controlling large and small-scale variability in ionospheric composition and density structures

The accomplishment of these nine prioritized science objectives for the TIMED mission will also enable the first quantitative baseline to be established against which future anthropogenic changes in the MLTI will be assessed.

Table 3-2

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Highest Priority Science Objectives for Alternative Mission Scenarios

Highest Priority Science Objective:	To investigate and quantitatively understand the processes responsible for the energy and momentum budgets of the mesosphere, lower-thermosphere/ionosphere (MLTI)

Table 3-3

Prioritization by Science Area for Alternative Mission Scenarios

Core Science:	Energetics and dynamics of the critical transition region between 60 and 180 km
Contributing Science Elements:	Chemistry Ionospheric structure and electrodynamics Coupling processes within the transition region, and with adjacent regions

Table 3-4

Given the above science objectives and considering the imposed cost constraint, the actual mission scenarios were developed through interaction between the Science Team and the Project Team as illustrated in Figure 3-6. This involved a number of mission elements such as operation and lifetime, orbital scenarios and instrumentation.

Early on in this planning phase it was recognized that the mission lifetime would have a significant impact on cost due to requirements on reliability. Since the primary phase of the TIMED Mission could be accomplished in a time span of 2 years (as shown in Figure 3-3),

INTERACTION BETWEEN SCIENCE TEAM AND PROJECT TEAM

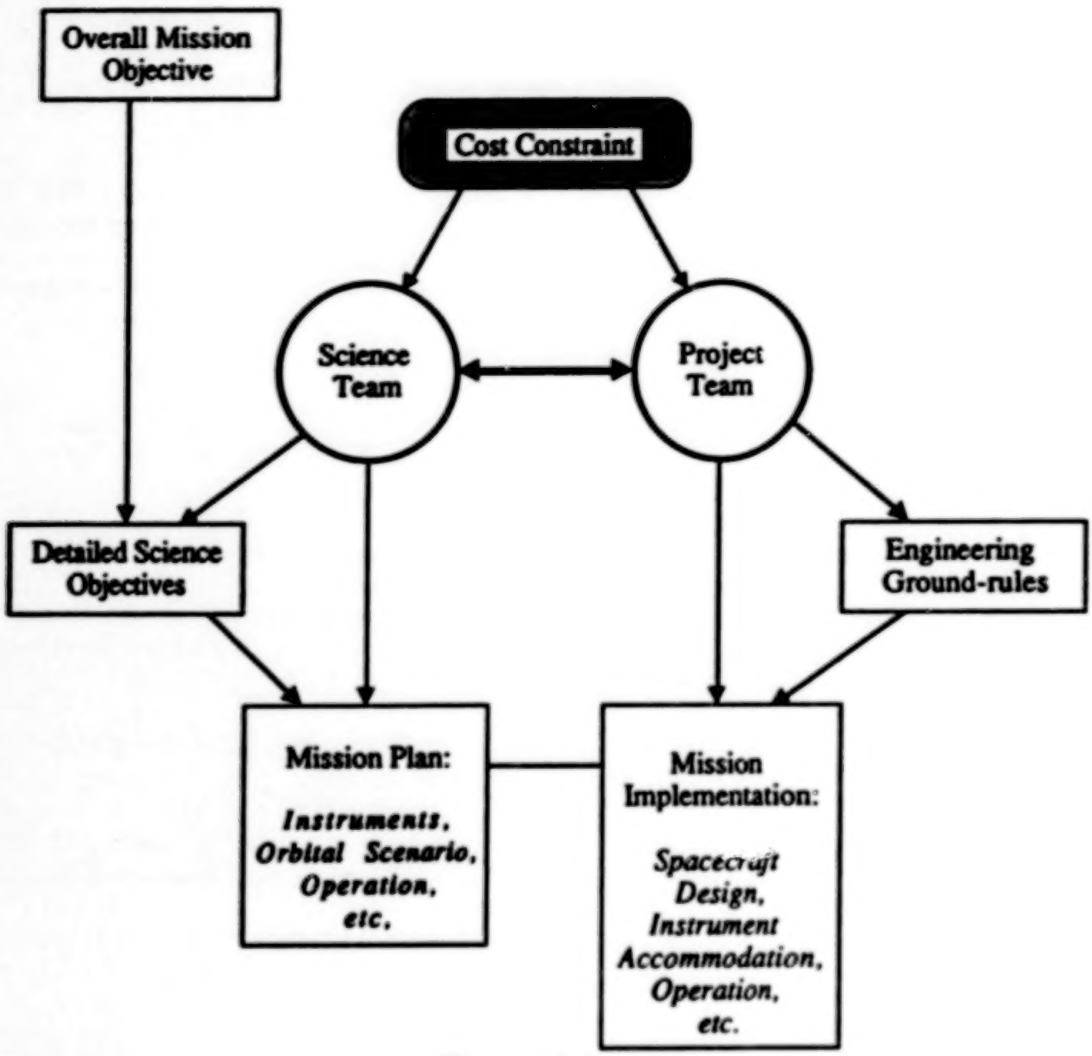

Figure 3-6

the SDT agreed to reduce the original nominal lifetime from 4 to 2 years, with the provision that the spacecraft would carry enough fuel for extended operation.

For the instrument payloads, the SDT took into account the factors listed below:

1) Importance for the core science objectives
2) Redundancy of measurements
3) Information provided by other scientific programs
4) Impact on spacecraft resources

The lower priority of objectives related to ionospheric structure and electrodynamics, relative to the core science of energetics and dynamics in the MLTI region, justified the removal of the boom instruments for measuring electric fields (EFD/PWE) and magnetic fields (MAG) which are demanding in terms of spacecraft resources. In the case of the EFD/PWE instruments, another justification was the partial overlap with the ion drift meter and retarding potential analyzer (IDM/RPA), which indirectly yields low resolution electric field measurements at altitudes above 160 km without complicating the spacecraft design.

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With the reduced science priority on ionospheric structure, the instrument for measuring auroral particles (EPS) was considered less important, in particular since some of the information can also be provided by the global UV airglow imager (GUVI). Moreover, National Oceanic and Atmospheric Administration (NOAA) and Department of Defense (DOD) spacecraft are carrying particle detectors that are expected to perform such measurements on an operational basis, albeit in different regions on the globe.

Another instrument considered for removal was the ion mass spectrometer (IMS) which can be justified based on the adopted science prioritization away from ionospheric structure, and on the overlap with the IDM/RPA instrument that measures the ionospheric density and the composition of the major ions. It was also argued that the instrument for measuring the neutral composition, winds and temperature (NMSWT) could be modified to provide the ion composition measurements.

In the area of the neutral density measurements, the accelerometer (ACC) was considered for removal since it cannot add much to the information already provided by the more potent NMSWT instrument.

In the area of remote sensing, the infrared limb sounder (IRLS) provides temperature measurements from 90 down to at least 60 km, both during day and night time, while the ultraviolet spectrometer (UVS) measures the temperature only during day time. Although the latter instrument performs other measurements as well, it was considered for removal in order to contain cost.

The instruments considered for removal, the rationale and science loss are summarized in Table 3-5.

Based on the above considerations for mission operation, orbital scenario and payload instrumentation, the SDT then defined the alternative mission scenarios given in Table 3-6.

The Option 0 is the two spacecraft mission with nominal lifetime of 4 years (allowing for extended operation beyond that). Its orbital scenario is shown in Figure 3-3, and the instrument payload is described in Table 3-1. This mission scenario was described in the SDT Report and was originally recommended to NASA for full implementation.

Option 1 is a two spacecraft mission, but with the nominal lifetime reduced from 4 to 2 years (allowing for extended operation beyond that). The instrument payload is the same as in Option 0, described in Table 3-1.

Option 2 compared to Option 1, reduces the payload by removing the instruments listed in Table 3-5.

Option 3 is similar to Option 2, except that the instrument and spacecraft requirements were relaxed because deep dipping (down to 130 km) was not required. For this option, it would be sufficient to fly the spacecraft (at least the one in high inclination) through the core region below 180 km altitude.

Option 4 is a two spacecraft mission which differs significantly from the other options. The spacecraft in low inclination orbit is the small explorer type with greatly reduced instrument payload and capabilities, carrying only three instruments for remote sensing: the imaging photometer (IP), the Fabry-Perot Interferometer (FPI) and the UVS. The demanding IRLS instrument, normally carried on TIMED-L, was moved to TIMED-H.

Since that instrument is considered important for the study of tides, the inclination for TIMED-H was reduced from 95 degrees to 70 degrees, which gives a precession rate sufficiently rapid enough to cover all local times and latitudes (below 70 degrees) in a period of about 65 days. The circular orbit inclination of 30 degrees was chosen to provide good local time coverage.

Option 5 is a one spacecraft mission in a 70 degrees inclination compromise orbit with a sequence of orbital phases adopted for TIMED-L in Options 1 through 3. It carries the entire instrument payload listed in Table 3-1.

Instruments Removed in Option 2

Instruments	Rationale and Science Loss
MAG Magnetometer	*Not core science* Loss of in-situ electric current information for Joule heating related to ionospheric structure and electrodynamics
EFD/PWE Electric-field Boom Instrument	*Not core science* Loss of in-situ vector electric field information mainly below ~160 km, where collisions are important; loss of ionospheric structure and electrodynamics science data
IMS Ion-Mass Spectrometer	*Not core science* Loss of in-situ spectral resolution and accuracy of ion composition related to ionospheric structure and electrodynamics
ACC Accelerometer	*Information provided by other instrument* (NMSWT) Loss of in-situ satellite drag measurements valuable at lowest altitudes
EPA Energetic Particle Analyzer	*Information provided by other instrument* (GUVI) *and programs* Loss of information on auroral energetic particles
UVS UV Spectrometer (TIMED-L only)	*Information provided by other instrument* Loss of ozone information

Table 3-5

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Proposed Alternative Mission Scenarios
(in order of decreasing science)

Option 0	Two-spacecraft TIMED mission, with 4-year nominal lifetime, and primary strawman instruments (described in Table 3-1).*
Option 1	Two-spacecraft mission, with mission lifetime reduced from 4 to 2 years, and primary strawman instruments (as in Option 0).
Option 2	Same as Option 1, but with strawman instruments removed (primarily those involving booms and providing redundancy with other instruments and/or programs).
Option 3	Two-spacecraft mission, with 2-year lifetime, and reduced strawman instrument payloads (as in Option 2), but with simplified spacecraft requirements because deep dipping not required.
Option 4	Two-spacecraft mission, with 2-year lifetime, without the requirement for deep dipping (as in Option 3). One TIMED spacecraft in intermediate inclination orbit (70 degrees). Second spacecraft based on a "small explorer satellite" design in low inclination orbit (30 degrees) with payload for remote sensing.
Option 5	One-spacecraft mission, with 2-year nominal lifetime, and all primary instruments (described in Table 3-1). Intermediate inclination orbit (70 degrees), initially circular followed by dipping.

*Originally recommended to NASA for full implementation (SDT Report, Volumes I and II, Reference 2)

Table 3-6

4.0 STRAWMAN INSTRUMENTS

As mentioned in the previous section, the Science Definition Team developed a strawman list of candidate instruments which they prioritized as part of the alternative mission scenarios. This gave the Project Team a list of instruments for each mission option. The data on the instruments was drawn from an extensive heritage of similar instruments that flew on such missions as the Orbiting Geophysical Observatory (OGO), the Atmospheric Explorers (AE), the Dynamics Explorer (DE), Nimbus, Solar Mesospheric Explorer (SME) and the Pioneer Venus Orbiter. Further information on the instruments, such as physical characteristics, resource requirements and operational considerations were gathered from discussions with the Science Team.

Once the Project Team got started, the data base was put under configuration control. This ensured that all the options were based on the same instrument data base. Some new information came in after the data base was frozen, but no changes were significant enough to invalidate the observatory design concept.

The final Phase A mission concept has 15 instruments, with 9 on TIMED-H and 6 on TIMED-L as shown in Table 4-1. All of the TIMED-L instruments are copies of instruments flown on TIMED-H, except for the Infrared Limb Sounder (IRLS), which is unique to TIMED-L. In other words, there are only 10 unique instruments that must be designed and developed. The other 5 are copies or duplicates.

Strawman Instrument Complements

Type	Instrument	Acronym	TIMED-H	TIMED-L
Remote Sensing	Fabry-Perot Interferometer	FPI	X	X
	Imaging Photometer	IP	X	X
	UV Spectrometer	UVS	X	-
	Near Infrared Spectrometer/ Photometer	NIRS/P	X	-
	Global UV Airglow Imager	GUVI	X	-
	Solar EUV Spectrometer	SEUVS	X	-
	Infrared Limb Sounder	IRLS	-	X
In-Situ	Neutral Mass Spectrometer/Wind and Temperature	NMS/WT	X	X
	Ion Drift Meter/Retarding Potential Analyzer	IDM/RFP	X	X
	Langmuir Probe	LP	X	X
		TOTAL	9	6

Table 4-1

Separate instrument data sheets for each instrument are provided in Appendix A. These sheets summarize the characteristics, operations and spacecraft subsystems requirements of each instrument. Tables 4-2 and 4-3 list the instrument complement for the TIMED-H and TIMED-L spacecraft and summarize the major instrument characteristics (size, weight, power, data, etc.) which was used in the Phase A Study for accommodation of the instruments.

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TIMED-H MISSION INSTRUMENTS AND CHARACTERISTICS

INSTRUMENTS	COMPONENT DIMENSIONS (CM)	VOLUME IN S/C (CM3)	WEIGHT (KG)	AVG. POWER (WATTS)	DATA RATE (KBPS)	POINTING DIRECTION	INSTR. FIELD OF VIEW (DEGREES)	CLEAR FIELD OF VIEW SEMI-CONE ANGLE (DEGREES)	INSTR. CUT-OUT IN SOLAR ARRAY (CM)
Fabry-Perot Interferometer	60x10x10 7x7x5	6,245	9.6	7.0	1.28	Limb 45° off Wake(2)	±4°V x 2°H (2)	45°	10x10
UV Spectrometer [1]	51x13x43	28,509	7.3	5.0	0.064	Limbb Wake	±2°V x 0.73°H	25°	13x18
Imaging Photometer	45x8x8 7x5x7	3,125	8.4	10.0	1.28	Limb Wake	±4°V x ±7.5°H	45°	8x8
Near Infrared Spectrometer/Photometer [2,3]	46x13x13	18,538	5.0	7.5	0.256	Limb Wake	3°V x 0.25°H	22.5°	13x6
Global UV Airglow Imager	40.6x17.8x20.3 55.9x20.3x25.4	43,404	15.0	22.0	11.0	Nadir	12° along track; 140° across track	20°	7.5x15
Neutral Mass Spectrometer/Wind and Temp.	35.5x17.8x27.9 17.8x17.8x10.2	20.862	12.0	24.0	1.536	Ram	180°	180°	17.8x10.2
Ion Drift Meter/Retarding Potential Analyzer [4,5]	21x25x23	12,075	8.0	8.0	2.2	Ram	60° semi-cone angle	60°	28x41
Langmuir Probe [6]	16x16x16 0.4 diax30 (2) 5x5x2 (2)	4,212	2.5	4.0	1.0	1 Sensor II to Zenith, 1 Sensor II to Nadir	30° semi-cone angle about ram	90°	5x5 (2) 2x2 (2)
Solar EUV Spectrograph [7]	40x20x40	32,000	11.0 inst. 7.0 plat.	7.5	0.4	Sun-pointing	1° semi-cone angle	1°	20x40
TOTALS		169,060	85.5	95.0	19.016				$0.28 M^2$

[1] Aligned with NIRS
[2] Requires 0.02 M^2 radiator
[3] Aligned with UVS
[4] Requires isolated ground plane 28x41 cm
[5] Aligned with NMS/WT
[6] Requires 2 probes 0.4 cm in diameter, each 30 cm long
[7] Mounted on Solar Pointing Platform

Table 4-2

TIMED-L MISSION INSTRUMENTS AND CHARACTERISTICS

INSTRUMENTS	COMPONENT DIMENSIONS (CM)	VOLUME IN S/C (CM^3)	WEIGHT (KG)	AVG. POWER (WATTS)	DATA RATE (KBPS)	POINTING DIRECTION	INSTRUMENT FIELD OF VIEW (DEGREES)	CLEAR FIELD OF VIEW SEMI-CONE ANGLE (DEGREES)	INSTRUMENT CUT-OUT IN SOLAR ARRAY (CM)
FABRY-PEROT INTERFEROMETER	60X10X10 7X7X5	6,245	9.6	7.0	1.28	LIMB 45° OFF WAKE (2)	±4°V X 2°H	45°	10X10
INFRARED LIMB SOUNDER[1]	93.9 RADIUS X 90° 61 WIDE	422,427	72.0	115	3.0	LIMB 40° OFF WAKE	±6.5°V X 1°H	35°V X 25°H	10X20
IMAGING PHOTOMETER	45X8X8 7X5X7	3,125	8.4	10.0	1.28	LIMB WAKE	±4°V X ±7.5°H	45°	8X8
NEUTRAL MASS SPECTROMETER/WIND AND TEMPERATURE	35.5X17.8X27.9 17.8X17.8X10.2	20,862	12.0	24.0	1.536	RAM	180°	180°	17.8X10.2
ION DRIFT METER/RETARDING POTENTIAL ANALYZER[2,3]	21X25X23	12,075	8.0	8.0	2.2	RAM	60° SEMI-CONE ANGLE	60°	28X41
LANGMUIR PROBE[4]	16X16X16 0.4 DIA.X 30 (2) 5X5X2 (2) 2X2X2 (2)	4,212	2.5	4.0	1.0	1 SENSOR ⊥ TO ZENITH, 1 SENSOR ⊥ TO NADIR	30° SEMI-CONE ANGLE ABOUT RAM	90°	5X5 (2) 2X2 (2)
TOTALS		468,946	112.5	188.0	10.296				0.18 M^2

[1] REQUIRES 0.69 M^2 THERMAL CONTROL RADIATOR SURFACE IN ANTI-SOLAR DIRECTION.
[2] REQUIRES ISOLATED GROUND PLANE 28 X 41 CM
[3] ALIGNED WITH NMS/WT
[4] REQUIRES 2 PROBES 0.4 CM IN DIAMETER, EACH 30 CM LONG

Table 4-3

The instrument operational scenarios were also investigated as part of this study. Table 4-4 is a summary of where and when the instruments should be operated for elliptical and circular orbits. Figure 4-1 shows pictorally when the TIMED-H instruments are operated during the elliptical phase of the mission. It shows how long they are operated and where in the orbit they are operated as a function of altitude versus time from apogee. A 25 percent duty cycle is shown as a dashed line when it is less than the desired duty cycle which is shown as a solid line.

TIMED INSTRUMENT OPERATIONAL SCENARIO*

INST.	ELLIPTICAL ORBIT	CIRCULAR ORBIT
FPI	Operate below 600 km	Operate continuously
UVS	Operate at ≤1500 km during daylight. During night: some calibration, off otherwise.	Operate during daylight. During night: some calibration, off otherwise.
IP	Operate below 900 km	Operate continuously
NIRS/P	Operate continuously at ≤1500 km	Operate continuously
GUVI	Operate continuously ≤500 km	Operate continuously
NMS/WT	Operate below 600 km. Standby mode above 600 km.	Operate below 600 km. Standby mode above 600 km.
IDM/RPA	Operate continuously at all altitudes	Operate continuously
LP	Operate with other in-situ instruments below 1000 km.	Operate continuously with other in-situ instruments
IRLS	N/A	Operate continuously at 500 km (450 km acceptable)
SEUVS	Operate ~ 10-15 min/orbit (minimum 2 min/orbit) as high as possible above 300 km pointing at sun	Operate ~ 10-15 min/orbit (minimum 2 min/orbit) anywhere in orbit pointing at sun

* UNLESS OTHERWISE STATED, INSTRUMENT OPERATION IS CONTINUOUS EXCEPT AS LIMITED BY DUTY CYCLE

Table 4-4

From a resource standpoint, most of the strawman instruments have fairly modest requirements. Power, weight, size and data rate are reasonable for this type of mission. The only tall pole is the IRLS which is large and requires considerably more power than the other instruments. The Project Team made some adjustments to the TIMED-L design to decouple the effect of the IRLS so that it did not drive the common spacecraft design. How this was done is discussed further in Section 5.0.

In terms of viewing requirements, the TIMED strawman instruments fall into four classes: ram pointers; limb scanners; [illegible]; and a sun pointer.

Some of the in-situ sensors must be pointed into the velocity vector when they are collecting science data. To accommodate this, the spacecraft normal control mode pitches the spacecraft through one revolution per orbit (rpo). This permits these instruments to look forward along the velocity vector (ram direction) all along the orbit and also permits other instruments to look in the zenith, wake, nadir or limb directions. For instance, the energetic particle detectors look upward and downward to examine precipitating and backscattered auroral electrons and ions and many of the remote sensors are pointed at the limb, but look at different angles with respect to the velocity vector.

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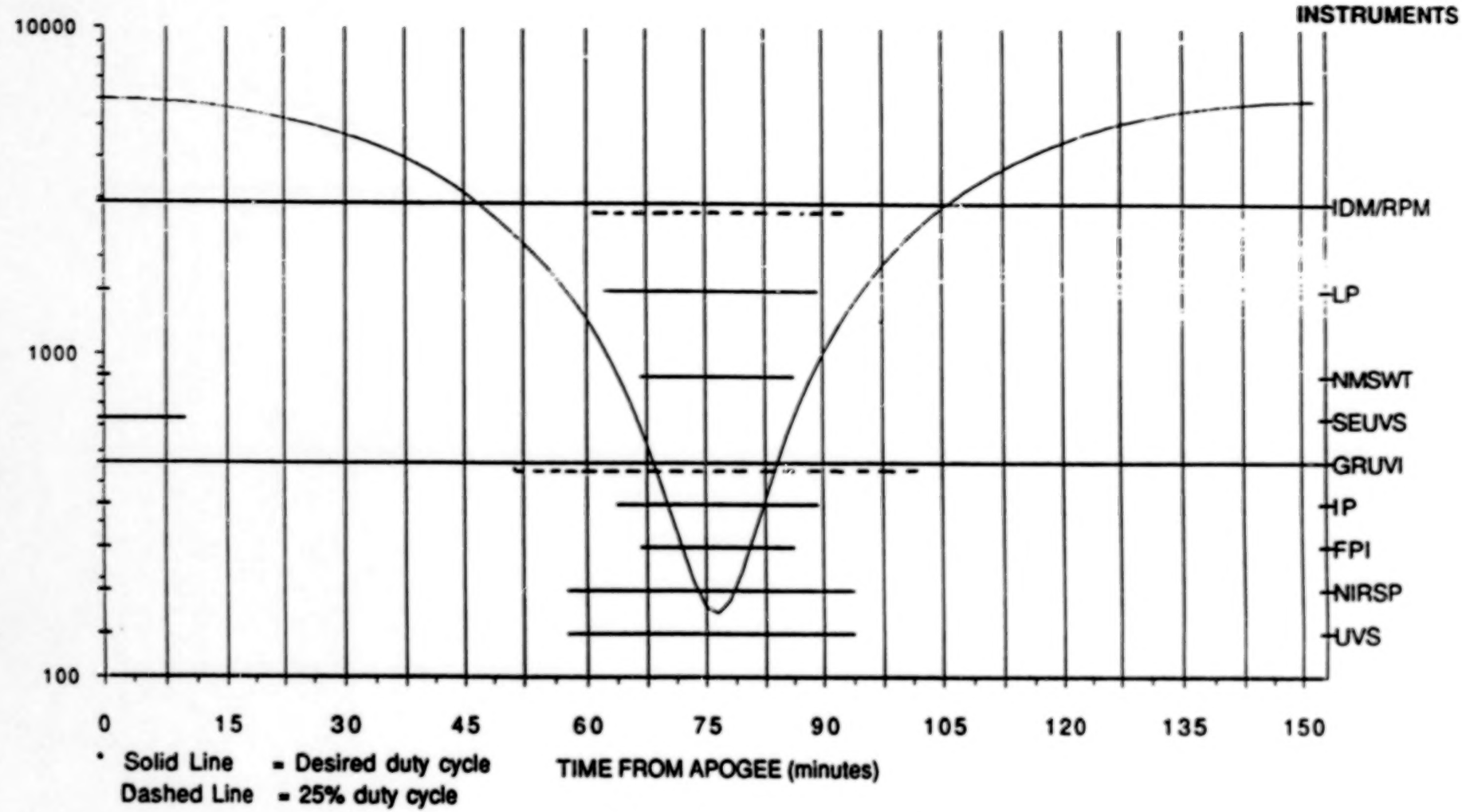
TIMED-H PHASE 1A INSTRUMENT OPERATIONS SCENARIO

Apogee = 6000

Perigee = 157



IDM/RPM: Ion Drift Meter/ Retarding Potential Analyzer
LP: Langmuir Probe
NMSWT: Neutral Mass Spectrometer/ Wind and Temperature

SEUVS: Solar EUV Spectrometer/ XUV Photometer
GRUVI: Global Remote UV Imager
IP: Imaging Photometer

FPI: Fabry-Perot Interferometer
NIRSP: Near-Infrared Spectrometer
UVS: UV Spectrometer

Figure 4-1

The ability to spin the spacecraft occasionally up to 4 rpm is also required to permit some instruments to calibrate.

A flat surface is required on the forward ram surface of the spacecraft to provide an unobstructed 2π-steradian field-of-view (FOV) for the forward facing instruments.

Table 4-5 presents the attitude control requirements and Table 4-6 presents the temperature limits for the instruments.

The conclusion of the Phase A Study left some concerns about the instrument accommodation that will be studied further in Phase B once the actual instruments have been selected. The radiation environment, instrument contamination, electromagnetic interference (EMI), attitude knowledge, thermal control and the yaw turn maneuver make up the list of concerns.

TIMED-L is in a low inclination, low altitude orbit which is fairly benign from a radiation standpoint. However, because TIMED-H is in a polar orbit that starts out with a high apogee, it will be passing through regions of high energy particles. Steps must be taken to ensure the survival of the spacecraft and instrument electronics in this environment. During Phase A, an investigation was conducted to quantify the TIMED-H radiation environment. The next step is to develop design guidelines that tell the instrument and spacecraft designers what they must do to minimize the effects of the radiation.

Another concern is instrument contamination. In addition to the normal cleanliness concerns during development and integration of the instruments, there are the on-orbit contaminants from the hydrazine thrusters and bombardment by atomic oxygen which can degrade the capabilities of the instruments. Studies must be conducted to quantify the effects of these contaminants and then establish guidelines to minimize whatever impacts they may have on the science.

Many of the instrument investigators expressed concerns that electric and magnetic fields generated by the spacecraft will interfere with their data collection. For example, some of the instruments will collect data on ions that are encountered by the spacecraft. The characteristics of those ions may be altered by the spacecraft fields unless care is taken to minimize EMI effects during the design of the observatory. The Dynamics Explorer Mission had to face these same problems ten years ago and some of their design guidelines were factored into the TIMED Phase A concept, but considerably more work needs to be done during Phase B to establish the specifics of the EMI plan.

In the area of attitude knowledge, it was recognized from the start that two of the strawman instruments had attitude knowledge requirements that were 0.05 degrees (1 sigma) (see Table 4-5) when the remainder of the instruments were satisfied with 0.1 degrees (1 sigma). It turns out the 0.1 degrees is on the border between relatively modest attitude sensors and an expensive set of sensors. Earth sensors and sun sensors can do an adequate job down to 0.1 degrees (1 sigma), but to go below that requires star trackers and even gyroscopes. It will also require a more stable structure and tighter alignment requirements. This was not in keeping with the spirit of the intermediate class mission. There were three alternatives: 1) Meet the tighter requirement for the entire observatory and add the cost; 2) Make the instruments that require the tighter knowledge provide it themselves with their own dedicated sensor; or 3) Re-examine the requirement with the Investigators' Working Group after instrument selection to see if the requirement can be relaxed. To keep costs down, the decision was made to stick with Earth sensors and sun sensors for Phase A and wait for the instrument selection before making the final decision in Phase B.

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TIMED INSTRUMENT ATTITUDE CONTROL REQUIREMENTS

INSTRUMENT	PLACEMENT	KNOWLEDGE	JITTER	STABILITY
FABRY-PEROT INTERFEROMETER	1°	0.1°	0.1°/0.25 SEC	0.1°/16 SEC
IMAGING PHOTOMETER	1°	0.1°	0.1°/0.25 SEC	0.1°/16 SEC
UV SPECTROMETER	1°	0.1°	0.007°/0.03 SEC	30"/12 SEC
NEAR INFRARED SPECTROMETER/PHOTOMETER	1°	0.1°	0.007°/0.03 SEC	20"/12 SEC
GLOBAL UV AIRGLOW IMAGER	0.5°	0.05°	1'/SEC	
SOLAR EUV SPECTROGRAPH[1]	10'	1'	10"/SEC	
INFRARED LIMB SOUNDER	1°	0.1° PITCH & YAW, 0.25° ROLL	±20"/SEC	±0.0075°/18 SEC
NEUTRAL MASS SPECTROMETER/WIND AND TEMPERATURE	2°	0.05°	0.05°/MIN	0.05°/MIN
ION DRIFT METER/RETARDING POTENTIAL ANALYZER	1°	0.1°	0.5°/SEC	
LANGMUIR PROBE	[illegible]°	1°	1°/SEC	1°/SEC

[1] MOUNTED ON SOLAR POINTING PLATFORM

Table 4-5

TIMED INSTRUMENT TEMPERATURE LIMITS

INSTRUMENT	OPERATING TEMPERATURE (°C)		NON-OPERATING TEMPERATURE (°C)	
	MAXIMUM	MINIMUM	MAXIMUM	MINIMUM
FABRY-PEROT INTERFEROMETER	40	0	60	-20
IMAGING PHOTOMETER	40	0	60	-20
UV SPECTROMETER	40	0	60	-20
NEAR INFRARED SPECTROMETER/PHOTOMETER*	40	0	60	-20
GLOBAL UV AIRGLOW IMAGER	35	-15	60	-30
SOLAR EUV SPECTROGRAPH	30	0	70	-50
INFRARED LIMB SOUNDER *	15	0	85	-45
NEUTRAL MASS SPECTROMETER/WIND AND TEMPERATURE	40	-20	60	-20
ION DRIFT METER/RETARDING POTENTIAL ANALYZER	50	-20	60	-30
LANGMUIR PROBE	50	-10	60	-10

* REQUIRES -5 DEG C HEAT SINK

Table 4-6

Stability and jitter were not fully covered during Phase A since most of the sources of disturbance could not be quantified. The effects of the momentum wheel and the IRLS cryogenic pumps were investigated, but mechanisms within the instruments that may be potential sources of disturbance were not studied.

A few of the strawman instruments have more stringent thermal control requirements than the rest of the instruments (see Table 4-6). It was decided that these instruments would have to be handled on a case by case basis during Phase B when the instrument designs are firmed up. Although it is not expected to be a major impact to the Phase A thermal design, the Phase B study will have to go to greater depth to tailor the thermal subsystem to meet requirements of those few instruments.

Another instrument concern is with the yaw turn maneuver which was developed as part of the Phase A concept as the most cost effective thermal system. Using a yaw turn to keep the sun on one side of the spacecraft gives the designers a dedicated cold end of the observatory. This design will be discussed further in Section 5.0. The result is that the ram facing instruments are turned to face the wake direction when the yaw turn is executed until the sun comes around to the other side of the observatory. This condition lasts for about a month and a half on TIMED-L and almost a year on TIMED-H. It was possible to choose launch conditions such that the ram instruments fly facing forward for almost the entire first year on TIMED-H, which is during the elliptical phase when those instruments are taking their best data. However, the subsequent yaw maneuver leaves the ram instruments flying backwards for about a year. Remote instruments and sun pointing instruments are not affected since the nadir direction and the sun direction are still maintained in body coordinates.

4.1 Fabry-Perot Interferometer (FPI) (TIMED-H and TIMED-L)

The FPI is a visible/near infrared limb scanning instrument with the necessary spectral resolution to determine the Doppler line profiles of various atmospheric emissions. The instrument is an etalon spectrometer with a baffled, twin-view, limb scanning system designed to enable horizontal vector wind profiles to be generated by using two near-orthogonal viewing directions.

The two FPI sensors are mounted to view the Earth's limb in the off-wake direction; one sensor is placed to view 45 degrees on one side of the wake direction, and the other sensor is placed to view 45 degrees on the other side of the wake direction. Both sensors are depressed 5 degrees to 16 degrees to view the limb.

The instrument contains a charged coupled device (CCD) array that operates at 200 degrees K and uses a thermal electric cooler (TEC) to cool the detectors. The TEC requires approximately 4 hours to stabilize; therefore, duty cycling of the cooler is not recommended even though the instrument may be duty cycled.

The FPI instrument is to be operated continuously at or below 600 km or as much as duty cycling permits. For circular orbits, operation during contiguous orbits is desired. For example, if the spacecraft makes 16 orbits per day and a 25 percent duty cycle is enforced, the instrument prefers to be on for 4 consecutive orbits and off for the next 12 consecutive orbits.

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4.2 Global Ultraviolet Imager (GUVI) (TIMED-H)

The GUVI is a spectrograph that uses a two-dimensional array detector and a scan mirror to image the atmosphere at selected UV wavelengths. It obtains horizon-to-horizon images when scanned across the track and a narrow slit is provided to permit limb scanning as well.

The instrument views the nadir direction from the wake side of the spacecraft and the sensor is pointed to image the auroral oval and the limb. The GUVI obtains horizon-to-horizon images by using a scan mirror or it can operate with the mirror in a fixed position. The instrument scans 140 degrees cross track from limb to limb in a 12 degrees swath along the track.

This instrument requires that there be no glint or reflected light from the sun and Earth off any booms or boom-mounted detectors.

The instrument is to be operated continuously for the elliptical and circular orbits or as much as duty cycling permits. If a 25 percent duty cycle is specified the instrument should be operated every orbit at altitudes between 300 and 3500 km (for orbits with 6000 km apogees) and at altitudes between 300 and 2000 km (for orbits with 3000 km apogees), or as much as duty cycling permits. Depending on the specific orbit parameters, the instrument may be operated occasionally outside the above maximums to acquire adequate latitudinal coverage. Operation below 300 km is possible, but depends on the exact shielding configuration.

The instrument can be turned completely off except for a low duty cycle 5 watt survival heater.

4.3 Imaging Photometer (IP) (TIMED-H AND TIMED-L)

The IP is a push-broom, filter-wheel, photometer operating in the visible and near infrared region of the spectrum. The photometric instrument breaks the horizontal field-of-view into a number of pixel elements that can be scanned vertically to provide a series of parallel altitude profiles of the mesospheric and lower- thermospheric airglow.

The IP images visual airglow emissions in a window that lies behind and below the spacecraft track. The instrument sensor is mounted to view the Earth's limb in the wake direction and is depressed 5 degrees to 16 degrees.

The IP instrument is to be operated continuously below 900 km or as much as duty cycling permits.

4.4 Infrared Limb Sounder (IRLS) (TIMED-L)

The IRLS scans the Earth limb 40 degrees off the wake direction. Limb scans can be repeated at 20 second intervals. In order to extend the long wavelength infrared measurements to the highest possible altitudes, the detector must be cooled cryogenically. This requirement is met in the strawman instrument by using a split Stirling cycle refrigerator, which gives the IRLS the largest power requirement of all the instruments.

A study was conducted to trade off the advantages and disadvantages of a mechanical cooler and a solid cryogen cooler. Both types of coolers have space flight heritage. The advantage of a mechanical cooler is that it has a longer life and can be turned off and on. Mechanical coolers are currently projected to have an operational life exceeding four years,

whereas solid cryogen coolers have a limited life and, by design, they cease to operate when all the cryogen has been vented. However, solid cryogen coolers are more reliable than mechanical coolers since the only operation is the opening of a valve on the vent line.

A disadvantage of the solid cryogen cooler is its weight and the fact that the weight changes. A system sized for two years would weigh 700 lbs. The weight itself is not so much a problem for TIMED as the shift in the observatory center of gravity as the cryogen is used up.

A disadvantage of the mechanical cooler is the power required to operate it and the potential vibration problem. The unbalanced force resulting from the operation of a single compressor has been measured to be approximately 10 lbs. on the British Aerospace mechanical cooler which is similar to the coolers flying on the UARS spacecraft. This force can be reduced by a factor of about 200 when two units are operated 180 degrees out of phase as shown in Figure 4-2. The Earth Observing System (EOS) mechanical cooler under development should be available as a flight design in late 1993. It has a life goal of 10 years with a minimum of 5 years. The cooling capacity is 0.8 watts at 80 degrees K at the end of life. It is designed for a low vibration of less than 0.05 lbs. force maximum with a 0.02 lb. force goal.

If a solid cryogen cooler were to be used, the preferred design would be a two stage, methane/ammonia cooler (see Figure 4-3). Methane has a much higher latent heat of sublimation when compared to nitrogen and argon, but has safety and handling problems. The ammonia is added as the secondary cryogen to reduce the weight. The secondary cryogen absorbs a large percentage of the parasitic heat before it can get to the methane. The latent heat of sublimation of ammonia is about 3 times that of methane. Of the total weight estimate of 700 lbs. for a two stage, methane/ammonia cooler for the IRLS, about 420 lbs. would be expendable cryogen.

For the Phase A Study, the mechanical cooler was baselined only to ensure that the spacecraft power resources were sized for a worst case instrument scenario. However, the solid cryogen cooler will be revisited in Phase B because of the potential savings in power.

The IRLS viewing configuration for this study was 40 degrees off the wake direction, depressed 18 degrees to view the Earth's limb, with the instrument located on the anti-sun side of the spacecraft. In this configuration, a 180 degrees yaw maneuver will point the instrument into the ram direction. To protect the optics from possible contamination and degradation, an aperture cover would close when the IRLS is pointed into the ram. A possible Phase B configuration would be to design the instrument to view 90 degrees to the velocity vector instead of 40 degrees off the wake. The 90 degrees will preclude the instrument from looking in the ram direction (in the yawed orientation) and allow maximum latitude coverage for a given orbit inclination.

Although the instantaneous FOV of the instrument at the limb varies as a function of orbit altitude, useful scientific data can be obtained for orbit altitudes between 1000 km and 300 km. In highly elliptical orbits, global coverage is difficult to achieve in any reasonable period of time. For this reason, the IRLS was put on TIMED-L so that it can operate in a circular orbit.

The instrument is relatively large (93.9 cm radius x 90 degree arc x 61 cm wide) and massive (72 kg). The two Stirling-cycle coolers drive the IRLS average power to 115 watts for normal operation and 44 watts for standby power. The instrument requires thermal control radiator surfaces in the anti-sun direction.

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TWO BAe COOLERS (BOTH OPERATING)

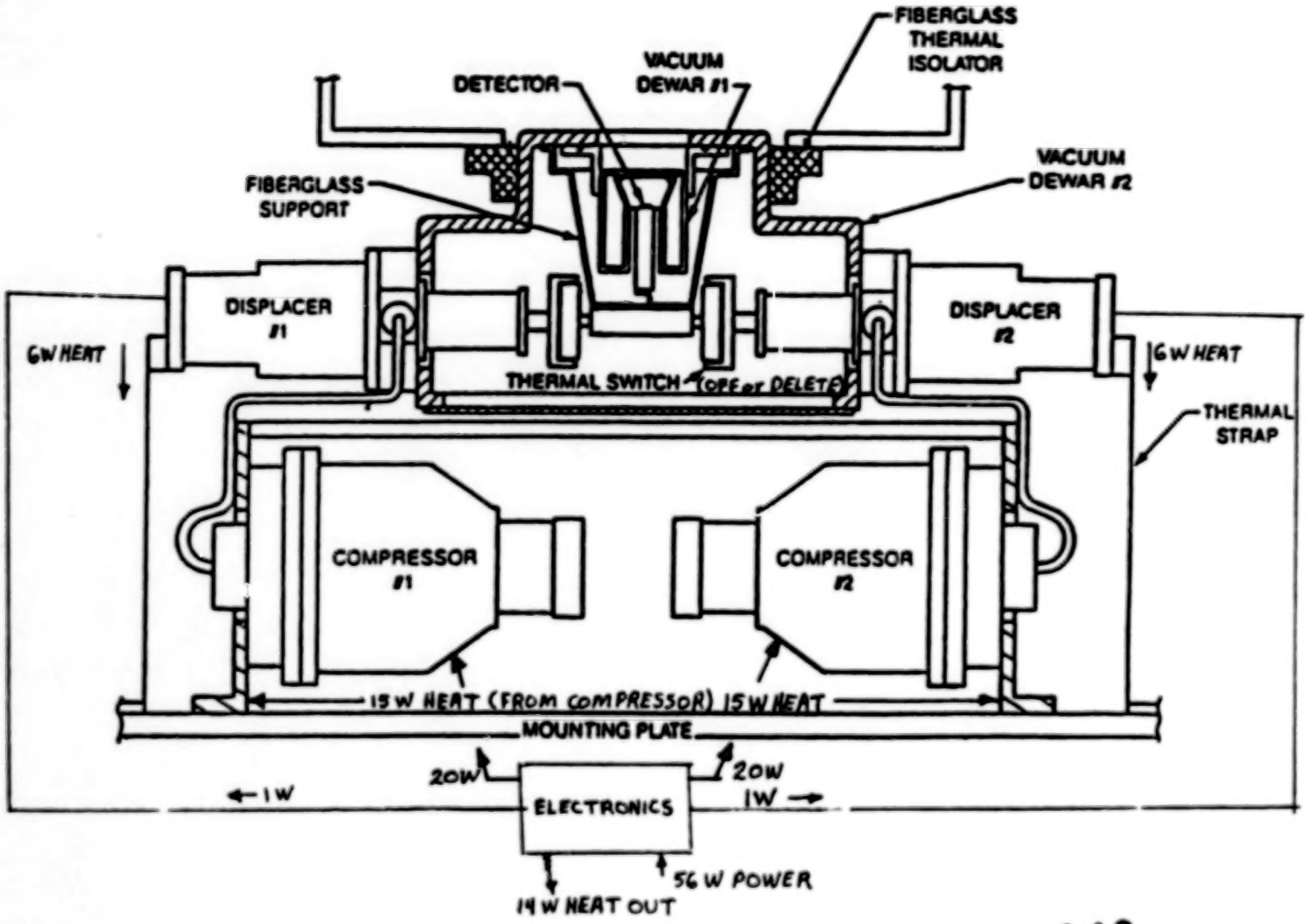

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Figure 4-2

TWO STAGE SOLID CRYOGEN COOLER

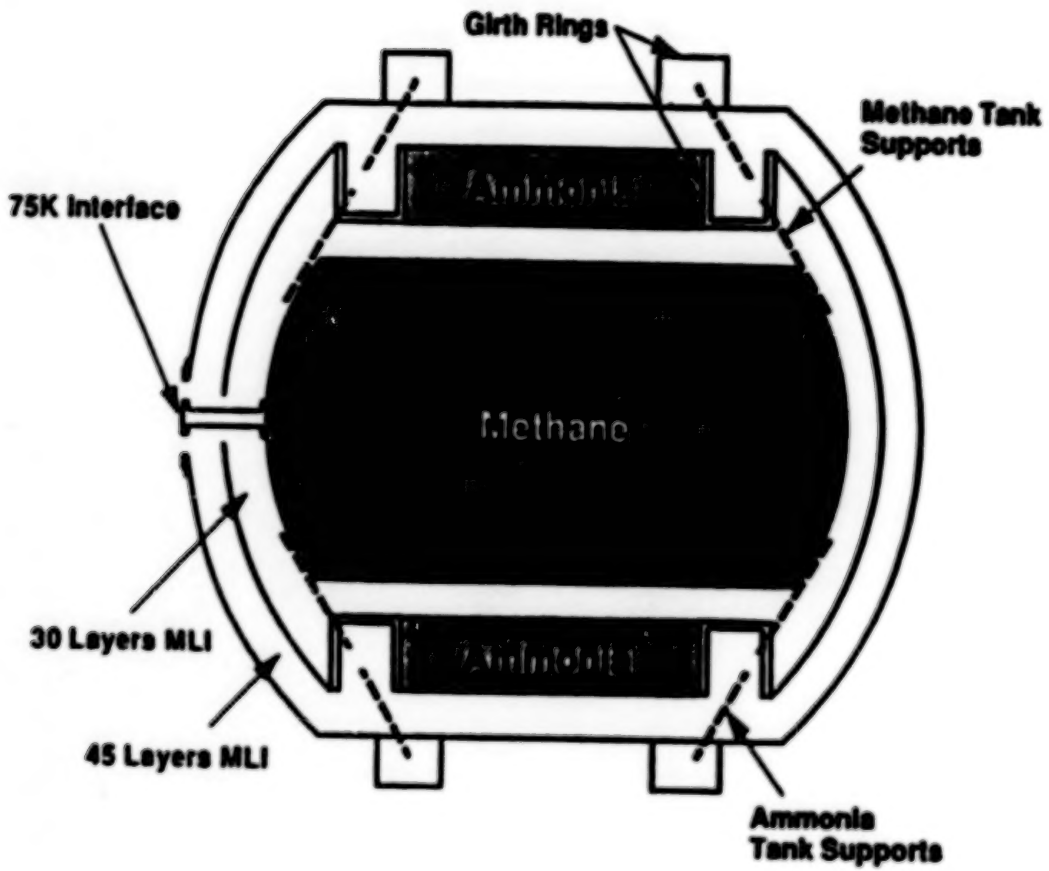

Figure 4-3

The IRLS instrument is to be operated continuously in circular orbits or as much as duty cycling permits. A circular orbit is by far the preferred orbit for the IRLS. If a 25 percent duty cycle is enforced, the preferred operation for the IRLS is to take data continuously for consecutive orbits (e.g., 4 orbits on per day). The instrument should not be turned off completely because of the long cool-down time using the mechanical refrigeration.

4.5 Near Infrared Spectrometer/Photometer (NIRS/P) (TIMED-H)

The NIRS/P is an imaging spectrometer which employs an array detector to provide for scanning in both wavelength and direction.

The NIRS/P is mounted to view the Earth's limb at 16.6 degrees below the wake direction. The instrument requires that there be no glint or reflected light from the sun and Earth by any booms or boom-mounted detectors.

The instrument requires a thermal control radiator surface area of 0.02 m^2 to dissipate 2.5 watts and it has a co-alignment requirement with the UV Spectrometer (UVS).

The NIRS/P instrument is to be operated continuously at or below 1500 km or as much as duty cycling permits.

4.6 Solar EUV Spectrometer (SEUVS) (TIMED-H)

The SEUVS combines an EUV spectrometer and a soft X-ray photometer to measure the full disk solar flux over the wavelength range from 50 to 2000 Angstroms. This instrument is mounted on a two-axis solar pointing platform (SPP) which points the instrument in solar azimuth and elevation relative to the spacecraft spin axis to an accuracy of 10 arc-min.

The instrument requires an unobstructed FOV on the sun-side of the spacecraft while making measurements and prefers a near-constant temperature.

The instrument has 3 aperture doors , a closed loop sun tracker, and microchannel plate (MCP)-array detectors that are somewhat sensitive to spacecraft magnetic field.

The SEUVS instrument is to be operated 10 to 15 minutes per orbit at as high an altitude as possible above 300 km while pointing at the sun. For circular orbits, the instrument should be operated 10 to 15 minutes per orbit anywhere in the orbit while pointing at the sun. This includes approximately 5 minutes for warm-up and possible slew-time for setting up solar observation. The instrument can be turned completely off for the remainder of the orbit. The instrument should never look in the ram direction and should be shut down during low-altitude (<200 km) portions of elliptical orbits. Generally, as a minimum, only one observing interval is required per day; however, turning the instrument off for long periods is undesirable. The duty cycle can be reduced, if necessary, by operating in alternate orbits.

4.7 Ultraviolet Spectrometer (UVS) (TIMED-H)

The UVS is a two-channel Ebert-Fastie spectrometer with a movable grating and a scan mirror to view the Earth's limb. The wavelength range of the instrument is from 250-340 nanometers. The UVS measures the altitude dependence of Rayleigh scattering. Ozone abundance is derived from the shape of the radiance curve in the spectral region of the

CAA

instrument. The Rayleigh-scattered profile enables neutral temperatures to be determined from approximately 50 to 90 km, with an accuracy of approximately 2 degrees K.

The UVS instrument is to be operated continuously (or as much as duty cycling permits) at or below 1500 km on the daylight portion of the orbit. The instrument can be turned-off in the eclipse portion of the orbit, except for some periodic calibration of the instrument.

4.8 Ion Drift Meter/Retarding Potential Analyzer (IDM/RPA) (TIMED-H and TIMED-L)

The IDM/RPA is mounted on the flat, forward-looking surface of the spacecraft normal to the velocity vector to measure ion velocities, temperatures and densities.

The RPA consists of a planar collector behind a series of fine grids whose potentials are adjusted to reject the ionospheric electrons and, in sunlight, to return photoelectrons to the collector plate.

The IDM has a split collector system which allows it to measure two mutually perpendicular angles of arrival of the thermal ions with respect to the sensor look direction. The instrument employs a four-segment collector behind double grids that are grounded to minimize transverse electric fields inside and outside the sensor.

In order to prevent stray fields from distorting the readings, the instrument requires a 28 x 41 cm ground plane flush with the spacecraft exterior surface, a conductive spacecraft, an isolated solar array, a positive ground for the solar array and power system and a magnetic field $\leq$0.01 gauss at the sensor apertures.

The instrument mounting requirements are for an unobstructed view in the ram direction and co-alignment with the NMS/WT.

The IDM/RPA instrument is to be operated continuously at all altitudes or as much as duty cycling permits. Assuming a 25 percent duty cycle, the instrument should be operated in elliptical orbits during all perigee passes. In circular orbits, the instrument should be operated daily through all equatorial passes, all polar passes, and for whole orbits when possible.

The primary operational mode is in the ram direction. Although the instrument takes no critical science data when the observatory is flipped because of a yaw turn, some wake measurements are desired for studies of subsonic plasma and spacecraft interactions.

4.9 Langmuir Probe (LP) (TIMED-H and TIMED-L)

The LP employs two cylindrical collectors mounted on the ends of two short booms located on opposite sides of the spacecraft which are perpendicular to the velocity vector. The collector potential is varied to generate a volt-ampere characteristic whose shape and amplitude are determined by the ambient electron temperature and density.

The booms are each 30 cm long and 0.4 cm in diameter. The collectors are mounted on the ends of the booms and are 5 cm in diameter. The preferred orientation is that one probe is parallel to the nadir direction and the other probe is parallel to the zenith direction. Other spacecraft locations are acceptable as long as the probes are perpendicular to the velocity vector and are on opposite sides of the spacecraft. The two probes will be pyrotechnically released and deployed in orbit.

The science requires that the LP operates simultaneously with the other in-situ instruments since each instrument provides only part of the information that is needed. In elliptical orbits, the LP instrument should be operated at altitudes below 1000 km. In circular orbits, the instrument should be operated continuously or as often as duty cycling permits.

To obtain a global snapshot from circular orbits (full latitude coverage), full orbit operations will be required, perhaps on every second, third, or fourth orbit, depending upon the available duty cycle at the time. When duty cycling, full latitudinal coverage will be possible only for specified parts of each consecutive orbit.

Assuming a 25 percent duty cycle, quarter orbit operations will be possible. Typical quarter orbits on a given day might be: 1) Equator to North Pole; 2) Equator to South Pole; and 3) 45 degrees N to 45 degrees S latitude across the Equator on either the nightside or the dayside, perhaps alternating between the two on alternate orbits or on alternate days.

4.10 Neutral Mass Spectrometer/Wind and Temperature (NMS/WT) (TIMED-H and TIMED-L)

The NMS/WT employs a mass spectrometer to measure the composition of the thermospheric neutral constituents, the neutral temperature, and three components of wind in the spacecraft frame of reference (ram, vertical and cross-track horizontal). The NMS/WT can employ either magnetic or RF separation of the ions of different species. Two rectangular baffles are moved repeatedly in front of the spectrometer to cast a wake upon its orifice.

The NMS/WT looks in the ram direction and requires a 2π steradian FOV. The instrument employs a metal and ceramic, break-off cover which is pyrotechnically actuated and separated in orbit.

The NMS/WT instrument is to be operated continuously in circular and elliptical orbits below 600 km, or as much as duty cycling permits, and should be in the standby mode above 600 km. The primary operational mode is in the ram direction with occasional operation in the spin mode to establish the background. Assuming a duty cycle of 25 percent, the instrument should be operated in elliptical orbits for the lowest 25 percent of the altitude range and in circular orbits for 1 out of every 4 orbits.

When the spacecraft is yawed around, the instrument does not take meaningful data in the reverse direction and may be turned off provided its temperature does not fall below the minimum specified.

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5.0 OBSERVATORY DESIGN

This section describes the observatory and subsystem designs. Although nine mission options were studied during Phase A, it is important to note that this report only addresses the final selected mission option. There is a brief discussion of the study results of the other mission options given in Section 5.12. Section 5.1 provides an overview of the observatory. Sections 5.2 through 5.10 describe the spacecraft subsystems. Section 5.11 summarizes the major spacecraft and instrument mass, power and data rate resources.

5.1 Observatory System

5.1.1 Requirements

The observatory satisfies the following requirements:

- Satisfy all science and mission requirements which include accommodating the strawman instrument complement itemized in Tables 4-2 and 4-3.
- Consist of two spacecraft with a mission lifetime and orbital profile shown in Figure 2-2.
- Survive and operate with possibly degraded performance after one failure.
- Accommodate a 25 percent instrument power duty cycle after 2 years at worst case sun orientation.
- Provide pointing knowledge of 0.3 degrees at the instrument boresight using ground processing and pointing accuracy of 1.0 degrees in all axes except during deep perigee passes through an atmospheric density of 6×10^{-9} Kg/M^3 where the roll disturbance may be up to 4 degrees.
- Consist of spacecraft designs which are essentially identical with low aerodynamic drag.
- Employ a protoflight hardware philosophy with designs which emphasize heritage.
- Incorporate the subsystem performance margins.
- Utilize the Wallops Ground Station for science data and TDRSS for health and safety plus tracking.

The MSIS-86 Thermosphere Model [Reference 6] is used for determining the propulsion subsystem required capability and attitude control subsystem performance.

Each subsystem design incorporates performance margins. The power subsystem accommodates the required orbital average power plus an additional 25 percent. The thermal control subsystem accommodates the additional power level. The command and data handling subsystem provides on-board storage for 24 hours of 100 percent instrument power duty cycle data plus 25 percent. The attitude control subsystem provides 100 percent margin in momentum wheel control authority and magnetic torque rod capacity (as compared to the capacity required to remove deep perigee disturbances within one orbit). The communications subsystem provides a 3 db minimum link margin. The propulsion subsystem provides sufficient propellant to accommodate the mission profiles with the margined observatory mass, including controlled re-entry. The structure subsystem maintains a 30 percent margin.

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5.1.2 Observatory Description

Each observatory is a nonogon (9-sided) cylinder configuration with an overall size of 2.1 meters in diameter and 1.7 meters in length and a total launch mass of 946 kg and 883 kg for TIMED-H and TIMED-L, respectively. Both observatories are nearly identical with the major exception of the instrument complement and the addition of a non-articulating solar array panel on TIMED-L and the addition of center-of-gravity compensators on TIMED-H.

Each observatory is modular in design. Figures 5-1 and 5-2 provide an overview configuration drawing of TIMED-H and TIMED-L, respectively. Subsystems are mounted on the lower mounting plate which also serves as a dedicated cold end radiator to remove internally generated heat. The propulsion module is located in the center of the observatory body with instruments mounted on the upper mounting plate. This design concept permits parallel assembly and testing during integration and test.

Figure 5-3 is the TIMED system block diagram showing major subsystems and interfaces. Figure 5-4 is the TIMED observatory coordinate system.

The power subsystem is a direct energy transfer system utilizing gallium arsenide solar cells and nickel cadmium batteries to provide a power capability of 215 and 283 watts orbital average during worst sun orientations.

A conventional momentum bias system provides observatory control about the pitch axis in all spinning modes. Momentum wheels provide some gyroscopic stability in both the roll and yaw axes and magnetic torque rods provide additional control and momentum unloading. In addition, center-of-gravity (cg) compensator mechanisms provide the ability to adjust the observatory center of gravity to center-of-pressure (cg/cp) offset to less than 1 centimeter.

The propulsion subsystem is a monopropellant blowdown system using metallic diaphragm tanks to provide a predictable cg with control of the cg/cp offset by selective propellant use. Thrusters provide orbit adjust and observatory yaw turn capability.

The thermal subsystem design features a dedicated, cold end radiator to maintain instrument and subsystem temperatures with heat pipes mounted to the external observatory shell to maintain acceptable solar array temperature.

The command and data handling subsystem implements the scheduling of science and spacecraft operations, manages on-board data storage using a solid state recorder, and using a MIL-STD-1773 data bus for instrument and subsystem communication, provides a cost effective and easily integrated platform. The flight software executing within the command and data handling subsystem supports Consultative Committee for Space Data Systems (CCSDS) communications protocols, fault detection and recovery of instruments and subsystems, and closed loop operations of the attitude and propulsion subsystems.

The communications subsystem, using only omni-directional antennas, implements command, downlink and ranging capabilities through both the Tracking and Data Relay Satellite System (TDRSS) and the Ground Station Tracking and Data Network (GSTDN), with wide-band telemetry downlink only through GSTDN.

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TIMED-H MODULAR OBSERVATORY

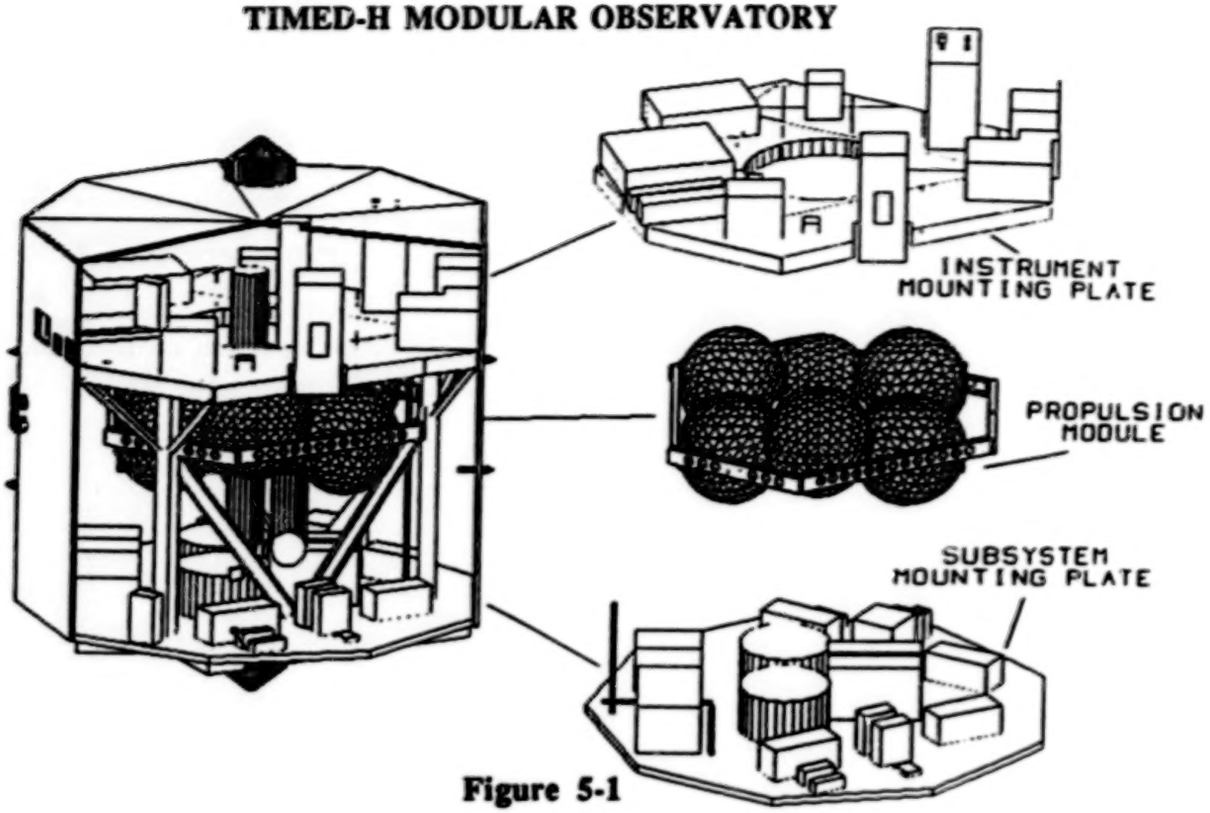

Figure 5-1

TIMED-L MODULAR OBSERVATORY

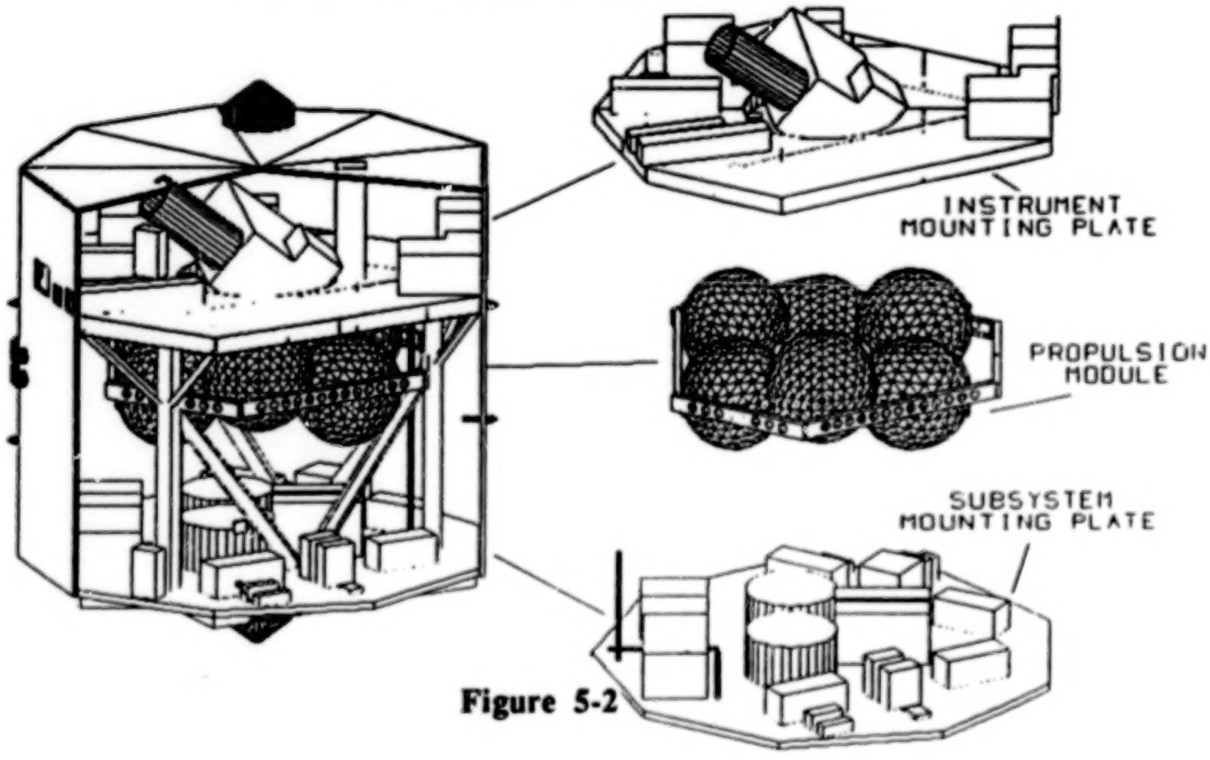

Figure 5-2

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TIMED SYSTEM BLOCK DIAGRAM

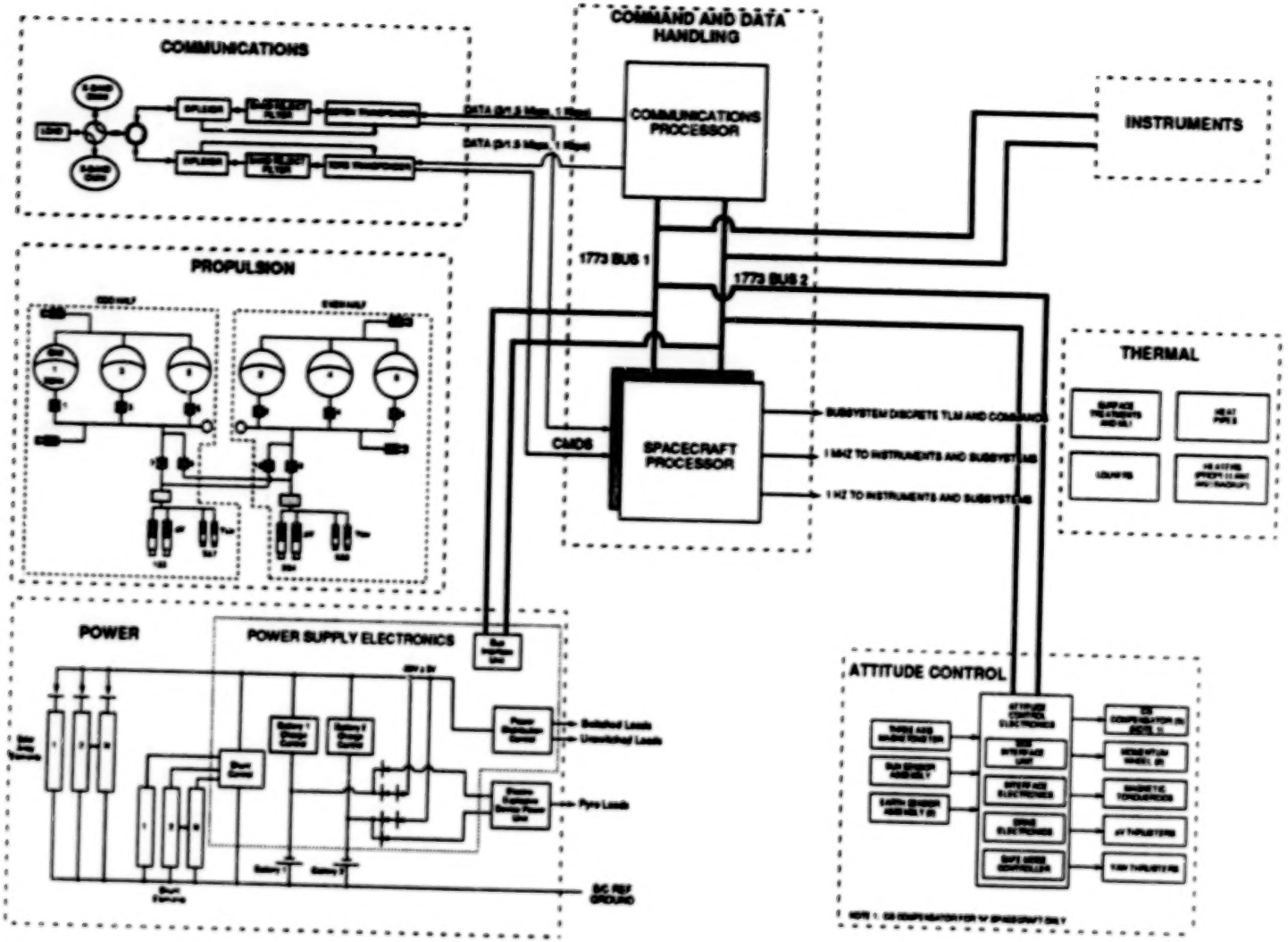
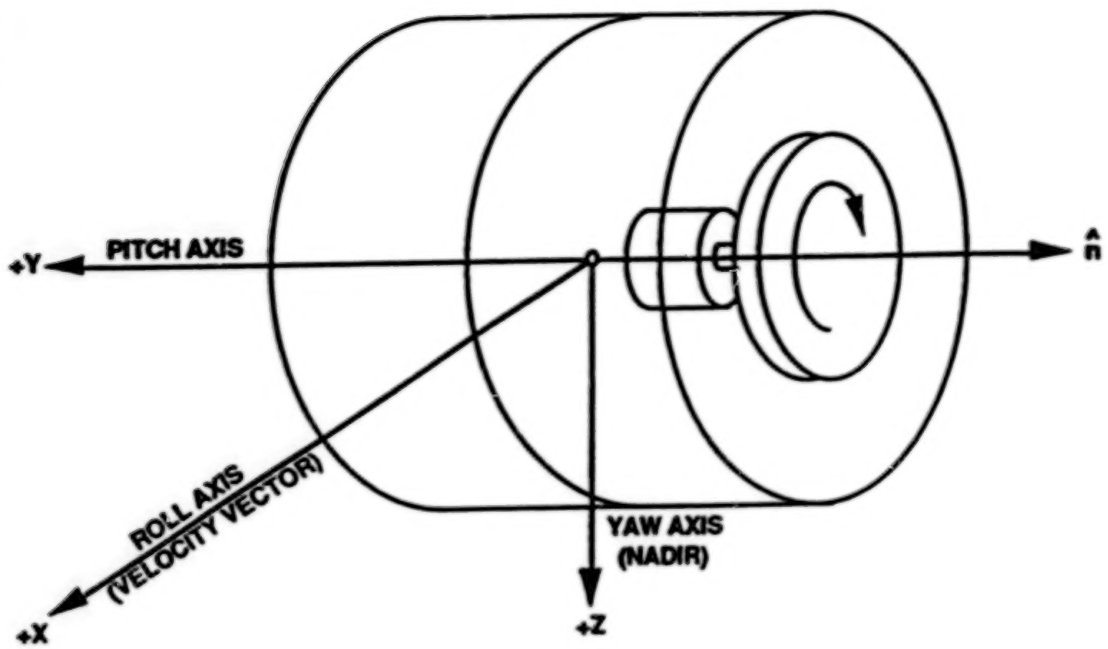


Figure 5-3

TIMED OBSERVATORY COORDINATE SYSTEM



Item	Axis System
Roll	X
Yaw	Z
Pitch	Right Hand About +Y
Velocity Vector	Along +X*
Nadir	Along +Z
Orbital Positive Normal	Along -Y

* Initial Mode - prior to 180 degree yaw maneuver required to maintain $\beta \leq 0$.

Figure 5-4

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5.2 Structure Subsystem

5.2.1 Requirements

The structure subsystem satisfies the following requirements:

- Provide a common structural design flexible enough to accommodate either instrument complement, including field of view (FOV) requirements.
- Survive launch loads
- Be compatible with a Delta II launch vehicle with a 2.89 m fairing and the 6019 Payload Attach Fitting (PAF).
- Provide center-of-gravity/center-of-pressure (cg/cp) offset of less than 1 centimeter.
- Provide sufficient surface area for side and end mounted solar cells and accommodate a 2.1 m^2 deployed solar array panel.
- Accommodate six, hydrazine tanks located around the observatory cg to support trimming of the observatory mass balance by the propulsion subsystem.
- Support thermal isolation of the body-mounted solar array from the internal spacecraft structure.
- Provide spacecraft design modularity.

5.2.2 Description

Each observatory's shape is a nine-sided cylinder, 2.1 m in diameter and 1.7 m in length. The modular structure design, shown in Figures 5-5 and 5-6 for TIMED-H and TIMED-L, respectively, consists of five primary elements described below.

Central Structure

The central structure is the primary load path between the Delta II launch vehicle 6019 PAF and the other elements. Its primary members are three longerons which extend from the PAF to the instrument mounting plate. These longerons have intermediate bolt interfaces, but this does not significantly sacrifice lateral stiffness performance. Radial members provide load paths to the circumferential extremes of the spacecraft. This central structure is an aluminum riveted structure with shear members connecting the three longerons. The design of these shear elements is driven by the mounting plate location. The central structure is open, allowing maximum radiative thermal coupling among elements.

Instrument Module

The instrument mounting plate, which is the primary structural element in the instrument module is an aluminum, isogrid structure which is reinforced locally to be compatible with load points. The mounting plate, which is 7.5 mm thick, does not extend to the full circumference in all facets and has cutouts in the center and edges to minimize weight and to accommodate an instrument. While these features may be detrimental to the stiffness of the platform, their effects are ameliorated by

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TIMED-H OBSERVATORY - EXPLODED VIEW

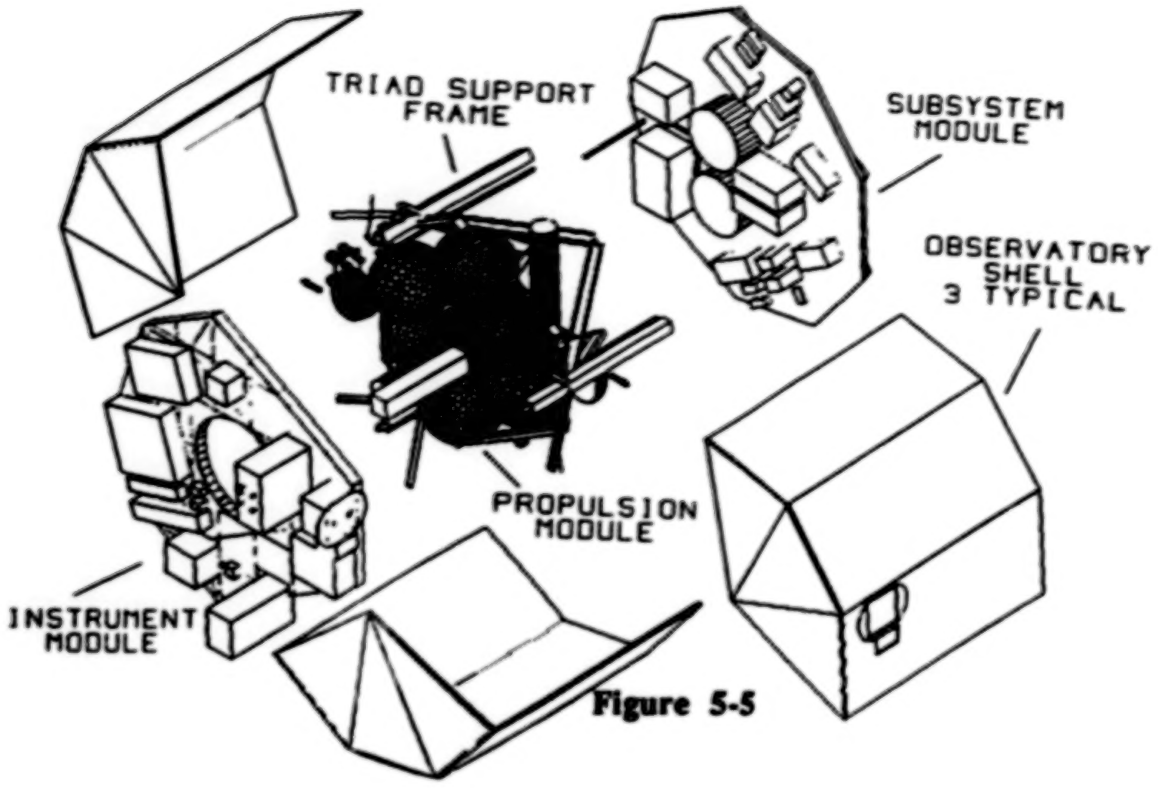

Figure 5-5

TIMED-L OBSERVATORY - EXPLODED VIEW

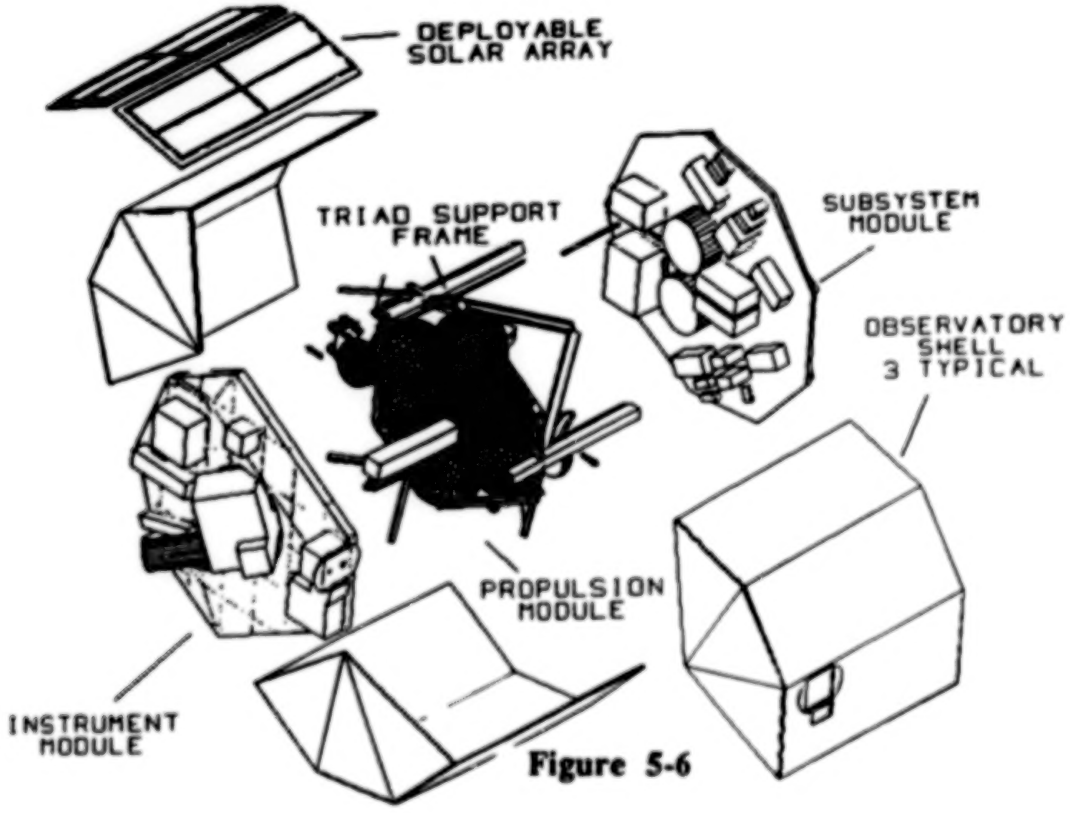

Figure 5-6

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the platform design. The isogrid construction provides a convenient, detachable interface for heat pipes and avoids compromising the modular nature of this structural element. These heat pipes conduct the instrument dissipated heat to the dedicated thermal radiator. Figures 5-7 and 5-8 depict the instrument layouts for TIMED-H and TIMED-L, respectively.

Solar Array Support Structure

The solar array support structure consists of a 1.25 cm aluminum honeycomb core with 1.0 mm aluminum facesheets. This support structure is thermally isolated from the spacecraft interior and has a network of heat pipes to maintain a low, near isothermal temperature for the solar cell panels. There are three, embedded heat pipes mounted on each facet. The heat pipes within the shell are thermally coupled by a central aluminum closure plate.

The support structure is integrated as three separate assemblies with each assembly consisting of three facets and three top segments. The three-segment assemblies are non-loadbearing.

Subsystem Module

The subsystem mounting plate, which is the primary structural element in the subsystem module is a honeycomb structure with a 2.5 cm core and 0.25 mm facesheets. The initial analysis indicates that an isogrid structure provides the required strength. Figure 5-9 shows the subsystem layout.

Propulsion Module

The primary structural elements for the propulsion module are inner and outer circumferential aluminum structures which provide the transition from the individual tanks back to the primary longerons. The tanks are located inside the primary longeron structure in order to reduce the observatory moments of inertia and locate major hardware masses close to the primary structure.

5.2.3 Performance and Analysis

Finite Element Model Analysis

A modal structural analysis was performed to determine the dynamic characteristics of the observatory configuration. The analysis shows compliance with the minimum frequency recommendations, 35 Hz axial and 10 Hz lateral, stated in the Delta II users manual [Reference 7].

Figure 5-10 is a representative plot of the finite element model (FEM) used to analyze the observatory structural design. The structural properties used in the analysis assumed 6061 T6 aluminum. The propulsion system was represented by nonstructural masses for the full tanks supported by the inner and outer circumferential members. The mounted electronic boxes were treated as distributed masses on the plates. The outer solar array shell was modeled as honeycomb material supported at its perimeter at the lower plates. No moment constraints were assumed at the PAF interface.

As shown in Table 5-1, the structural design has a first lateral mode at 21.6 Hz and a first axial mode at 37.3. These frequencies meet the recommended minimum natural frequencies.

The requirement to adjust the cg/cp offsets to less than 1 centimeter presents several problems with the TIMED-H configuration. To move the cg of this configuration to the geometric center of the spacecraft would require significant balance mass. Alternative structural configurations improve the ease of balancing the observatory at the expense of reducing design modularity.

TIMED-H INSTRUMENT LAYOUT

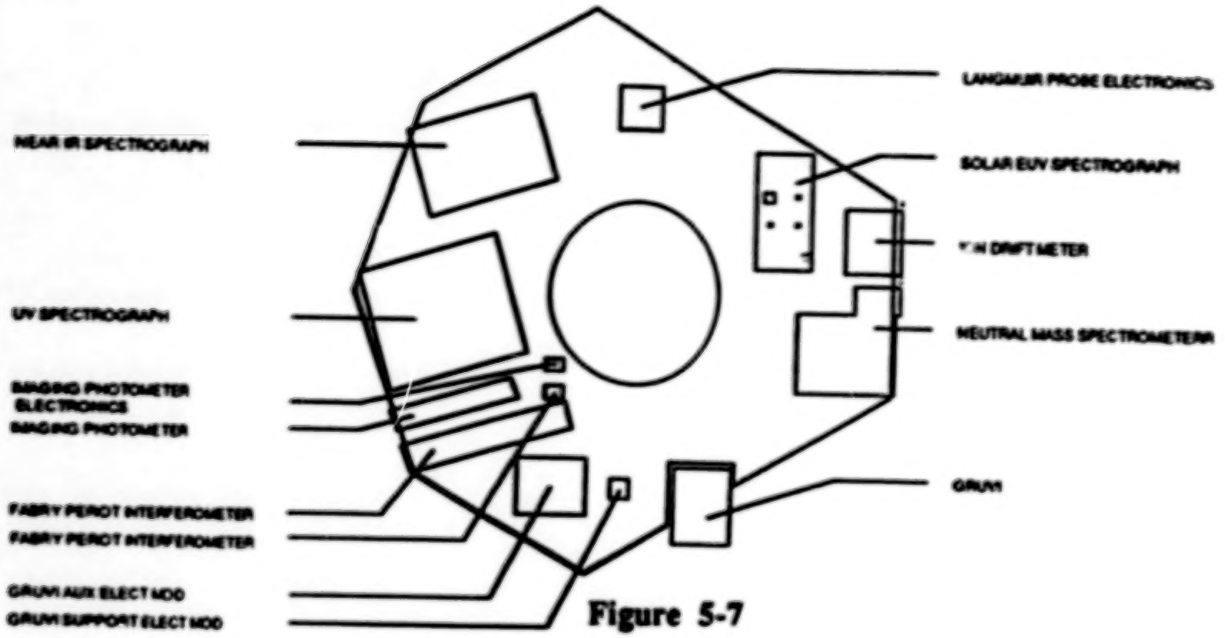

Figure 5-7

TIMED-L INSTRUMENT LAYOUT

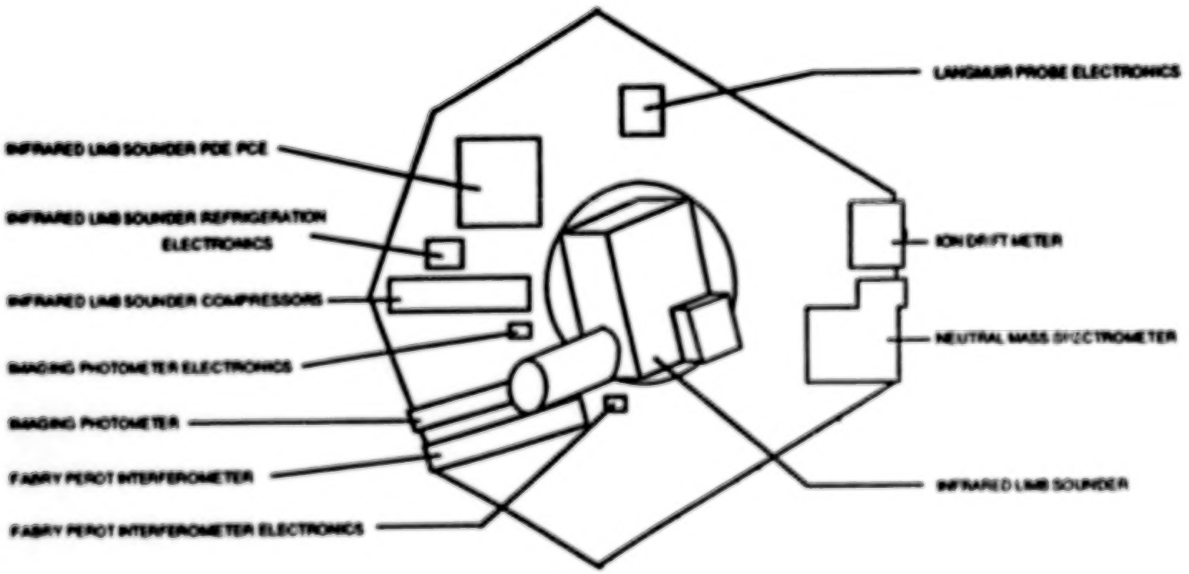

Figure 5-8

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TIMED SUBSYSTEM LAYOUT

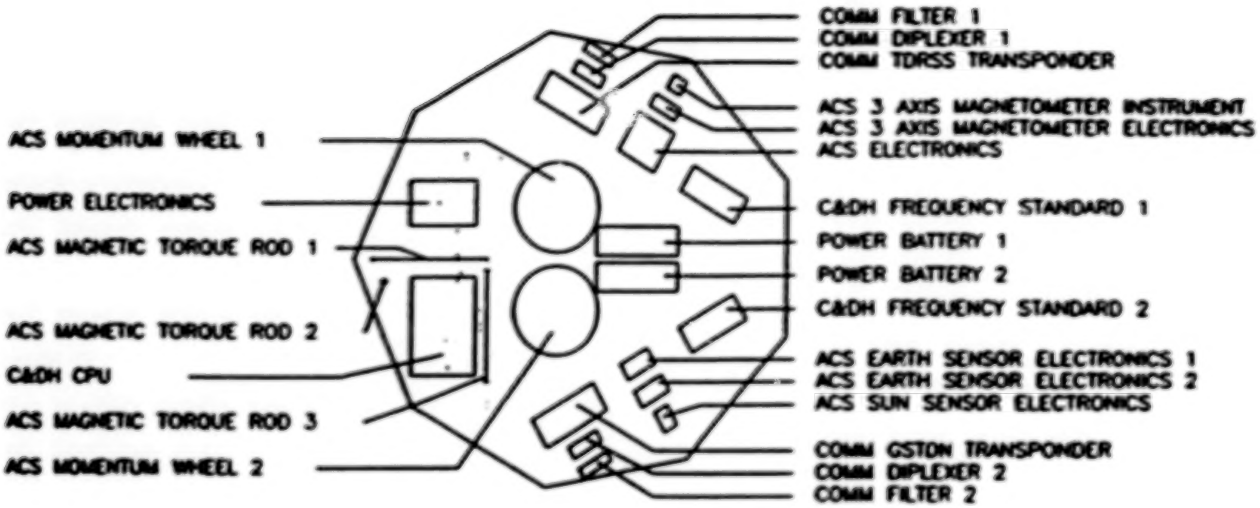

Figure 5-9

TIMED STRUCTURE FINITE ELEMENT MODEL

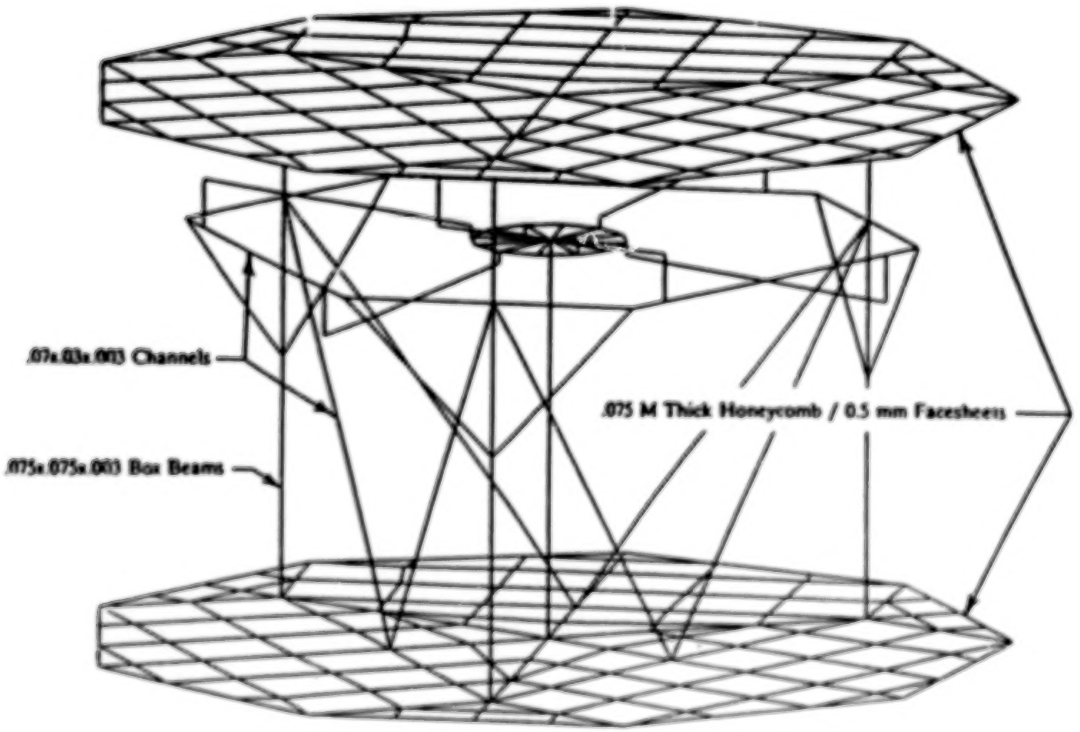

Figure 5-10

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An alternate configuration has the instrument mounting plate inverted and moved axially towards the top of the outer shell structure. This configuration was proposed to improve the cg/cp offset characteristics of the baseline design. To dynamically balance this configuration about the Z-axis of the spacecraft, using the maximum envelope of the spacecraft, would require 30 kg. Preliminary results show a first lateral mode at 21.5 Hz and a first axial mode at 37.4 Hz.

TIMED BASELINE STRUCTURE NORMAL MODES

MODAL EFFECTIVE WEIGHTS (kg)							
MODE	FREQ	X	Y	Z	RX	RY	RZ
1	21.55	3.49	521.22	0.00	546.08	3.77	0.00
2	21.55	521.00	3.50	0.00	3.78	546.10	0.00
3	24.70	1.38	34.04	0.00	227.63	8.44	0.00
4	24.70	34.04	1.39	0.00	8.44	227.63	0.00
5	27.16	0.00	0.00	38.61	0.00	0.00	0.00
6	30.36	0.00	0.00	0.00	0.00	0.00	140.74
7	34.66	18.70	0.04	0.00	0.04	101.82	0.00
8	34.66	0.04	18.70	0.00	101.81	0.04	0.00
9	37.29	0.00	0.00	169.64	0.00	0.00	0.00
10	49.87	0.00	0.00	456.68	0.00	0.00	0.00
11	49.96	1.73	3.75	0.00	5.78	2.04	0.00
12	50.00	3.75	1.73	0.00	2.04	5.77	0.00

Table 5-1

Instrument FOV Accommodations

The Aries CAD model was used to study the accommodation of the FOV requirements of the selected instruments and subsystems. Figure 5-11 provides illustrative examples of the FOV analysis.

No compromises were necessary to accommodate the FOV requirements for the instruments and subsystems on either observatory.

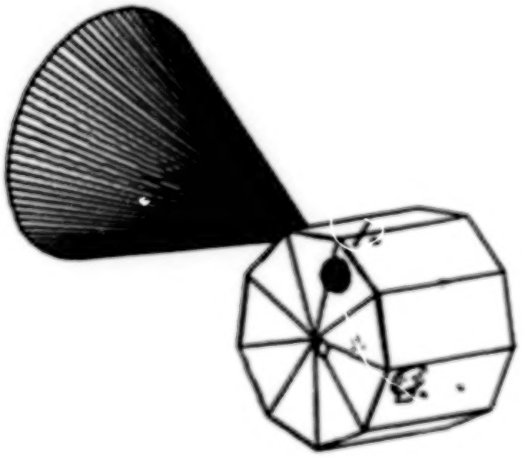
5.2.4 Reliability

The survivability factor for the structure is estimated to exceed 0.999. Electronic boxes are designed to sustain a single mounting bolt failure without stresses in the remaining mounting bolts exceeding ultimate values which potentially may result in yielding and the attendant misalignment. For structural systems that have extremely tight alignment tolerances, microyield stress factors are

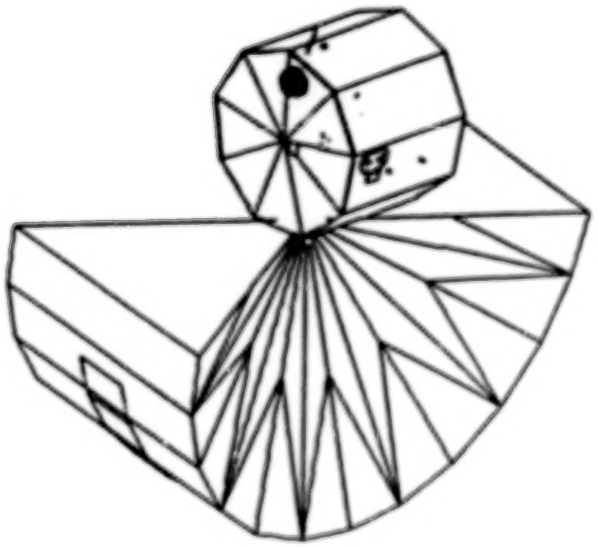
used to assess the integrity of the structure. These are essentially derated structural properties that ensure minimal yield under worst case loads.

The proposed structure design has significant heritage with spacecraft launched on a Delta II launch vehicle using a 6019 PAF and utilizes aluminum material and standard construction techniques.

TIMED FOV ANALYSIS ILLUSTRATIVE EXAMPLE



Near IR Spectrometer Instrument



GUVI Instrument

Figure 5-11

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5.3 Power Subsystem

5.3.1 Requirements

The power subsystem satisfies the following requirements:

- Provide 215 and 283 watts orbital average power requirements for TIMED-H and TIMED-L, respectively, for all beta angles after 2 years.
- Accommodate the instrument sprint mode, defined as a 100 percent instrument duty cycle for one full orbit, followed by 3 orbits for battery recharging with instruments not powered.
- Provide a negative 28 volt DC, unregulated bus for primary power to instruments and subsystems.
- Provide autonomous bus and battery fault protection.
- Provide two-fault tolerant electro-explosive device (EED) control for instrument boom deployment, solar array deployment, or instrument cover removal or any other hazardous operation.

5.3.2 Subsystem Description

The power subsystem is a direct energy transfer (DET) system that provides an unregulated, negative 28 ±5 VDC bus as shown in Figure 5-12. This system architecture provides solar array power directly to the loads. Bus voltage ranges between -23 and -33 VDC at the output of the power supply electronics distribution unit depending on orbit period and battery state of charge.

The DET system offers advantages in weight savings, higher efficiency, and design simplicity as compared to a regulated bus.

The power subsystem consists of the power supply electronics, the solar array, shunt dissipator, batteries and the EED power unit.

Each major subsystem component is described below in more detail.

Power Supply Electronics

The power supply electronics (PSE) unit contains the shunt controller which limits the bus voltage during the sunlit phase to -33 VDC, two individual battery chargers, bus and battery under/over voltage protection circuitry, and control and monitoring circuitry for power distribution and load switching circuitry.

Solar Array

The solar array uses gallium-arsenide on germanium substrate (GaAs/Ge) solar cells with a 18.3 percent cell efficiency at 28 degrees C. For radiation protection, fused silica cell coverglass 30 mil thick is used. The cells are electrically interconnected into series-parallel groups to provide the required voltage and the redundancy necessary for graceful degradation in case of interconnect failures or cell breakage. These groups of cells are bonded to the mechanical substrates with an insulating sheet of micaply in between. The cell groups are isolated from each other and from the remainder of the power subsystem with diodes to prevent power loss during eclipse or in case of shadowing or shorting. The side-mounted solar assembly consisting of all assembled array

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TIMED SUBSYSTEM BLOCK DIAGRAM

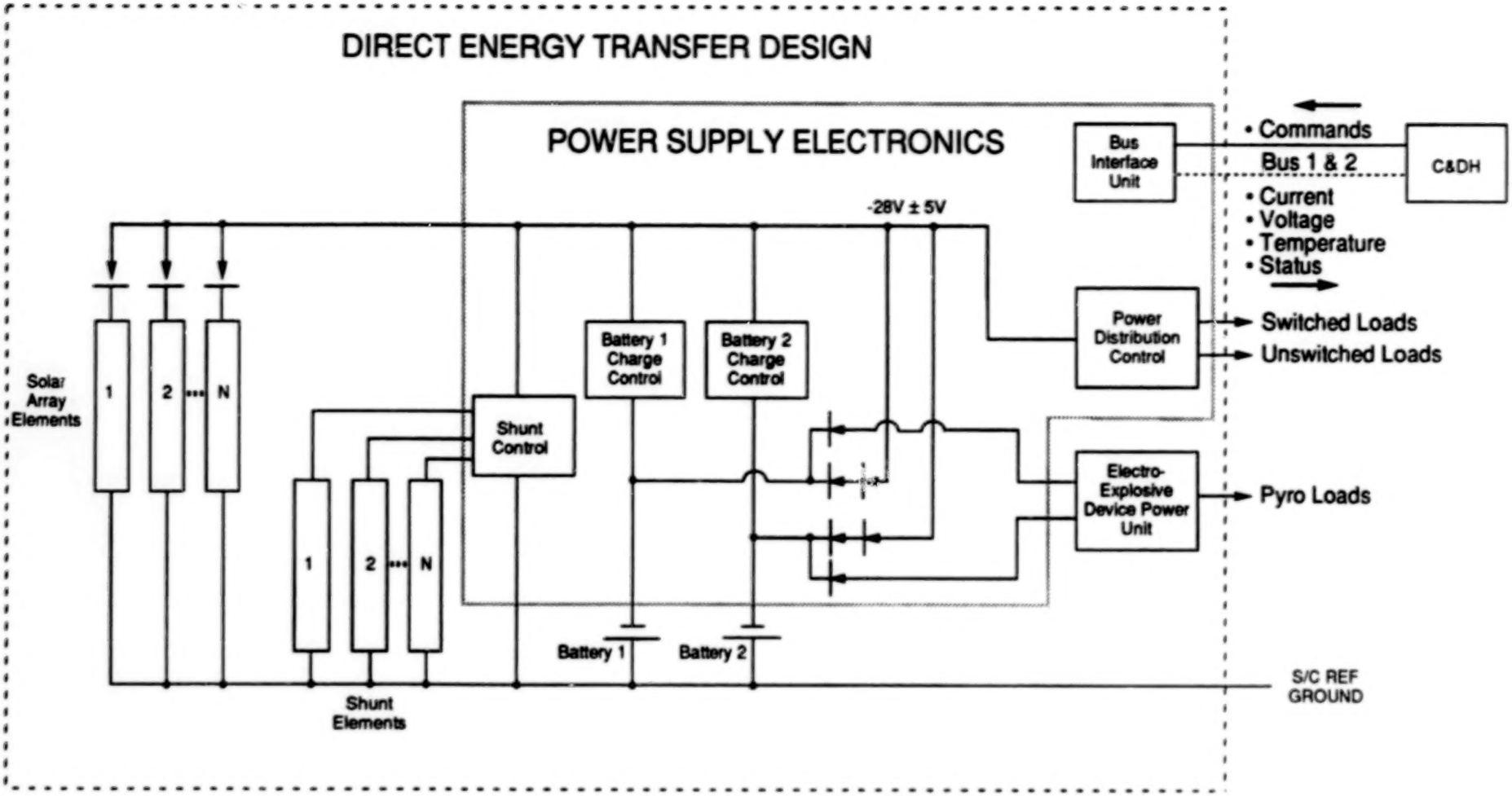

Figure 5-12

components and materials, excluding the mechanical substrate, weighs 3.33 kg/m^2 while the deployable solar assembly weighs 3.51 kg/m^2 because of a higher packing factor.

The TIMED-L, deployable, solar array is a spring-loaded, hinged panel utilizing a wrap-around panel configuration, 1.5 m x 1.4 m in area when fully deployed. The array panel is two aluminum honeycomb substrates with a core thickness of 1.25 cm and with aluminum facesheets of 0.25 mm thickness. The array is stowed on the side of the TIMED-L observatory and rotates 135 degrees to the operational position. Launch restraints for the deployable array are located on the instrument module.

Positioning the array on apex symmetrical panels simplifies all aspects of the design and implementation of this hardware.

Since all hinges of the array are spring driven, all deployment motions begin upon panel release. The erection of the panels to form a flat solar array involves only minor rotations and is accomplished using well-proven hinge/latch concepts developed on programs such as the Cosmic Background Explorer (COBE). The main rotation of the array to 135 degrees is the major deployment. This deployment is assisted by designing the array around a central, integral mast that interfaces with the spring motor drive located at the apex attachment point. The spring motor is a negator spring to avoid unnecessary acceleration of the array. A viscous damper will be incorporated with the spring motor to control the deployment rate. The solar array deployment sequence is shown in Figure 5-13.

TIMED-L SOLAR ARRAY STORED AND DEPLOYED CONFIGURATION

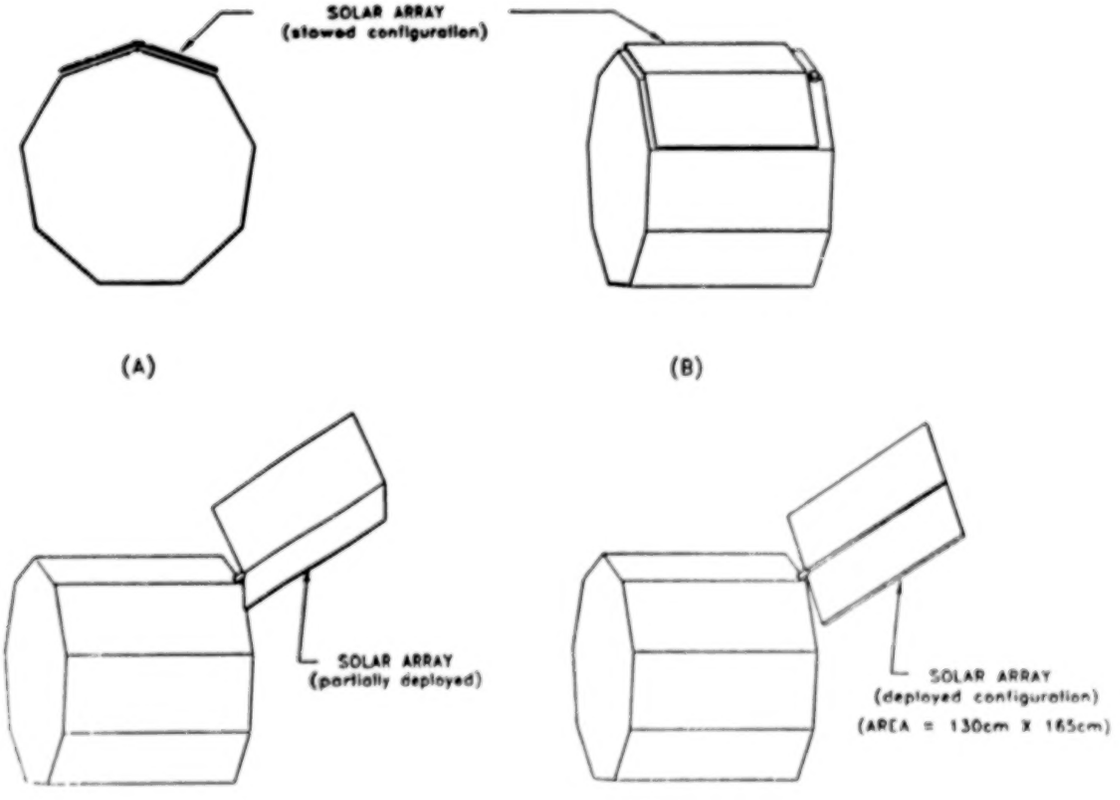

Figure 5-13

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For TIMED-H and TIMED-L, the total solar array area is 12.16 m^2 with 10.66 m^2 on the side section and 1.5 m^2 on the end section of the observatory body. In addition, TIMED-L has a 2.03 m^2 deployed solar array.

The side section of the solar array for TIMED-H generates 512 watts BOL, and 430 watts EOL, and for TIMED-L generates 475 watts at end of life (EOL) at beta 0 degrees. The end section of the solar array delivers 252 watts at beginning of life (BOL) and 212 watts EOL for TIMED-H, and 234 watts EOL for TIMED-L at beta 90 degrees. The deployed solar array on TIMED-L produces 128 watts BOL and 119 watts EOL average power at beta 0 degrees. Power generation capability is based on panel temperatures of 60 degrees C for the side section and 70 degrees C for the end section and deployed array. The total 1 million electron volts (MEV) equivalent electron fluence was calculated to be 1.7×10^{14} e/cm^2 and 1.27×10^{13} e/cm^2 for TIMED-H and TIMED-L, respectively.

Instrument and subsystem aperture cutouts occupy 0.3 m^2 of the side section and 0.1 m^2 on the end section of the observatory body. The cutouts are assumed to be spread out evenly over the surface.

Shunt Dissipator

The shunt dissipator limits the bus voltage to -34 VDC. It consists of externally mounted multiple strip heaters which are mounted on the rear of the solar panels. They are turned on sequentially as the available power exceeds the demand so as to maintain the bus at the required voltage limit during the sunlight phase of the orbit. The shunt dissipator design, to dissipate the maximum solar array power less the essential load power, is approximately 650 watts.

Battery

The power subsystem energy storage device consists of two, 22 cell, 20 ampere-hour, high performance super nickel cadmium (NiCd) batteries. High performance super NiCd batteries are selected because of their higher depth-of-discharge (40 percent) capability compared to regular NiCd batteries (25 percent) for the required cycle life between 0 degrees and 25 degrees. The battery depth of discharge is limited to 30 percent to ensure meeting the mission life requirements.

Electro-Explosive Device Power Unit

The EED power unit provides power switching to initiate pyrotechnic devices. Power to the EED power unit is derived directly from the NiCd batteries.

5.3.3 Performance And Analysis

Power subsystem components are sized by using the energy balance equation, which relates the orbital average spacecraft and instrument duty cycle power requirements, subsystem component losses and orbital parameters to the required solar array power, as shown below:

$$Psa \times H = PLs + PLe \times (Te/Ts)/(D1 \times D1 \times B \times C)$$

Where Psa is the required solar array power, PLs is the load power during sunlight, PLe is the load power during eclipse, and Te/Ts is the ratio of eclipse to sunlight time. The subsystem component losses are assumed as shown below:

Harness efficiency (H)	0.99
Average battery efficiency (B)	0.75
Charge controller efficiency (C)	0.90
Diode efficiency @ 26.8 V (D1)	0.97

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To compute the required solar array area, the variation in beta angle vs. days from launch, shown in Figures 5-14 and 5-15, result in variation in available solar array power. In addition, array losses, uncertainty and degradation due to manufacturing, radiation, ultra-violet and thermal cycling must be accounted for. The array losses are assumed as shown below:

Glassing	0.985
Cell mismatch	0.98
Cleanliness	0.99
Calibration error	0.98
Packing factor	0.85
Diode/harness	0.96
Maximum power voltage offset	0.93
Thermal cycling	1.00 (initial) minus 0.01 per year
UV degradation	0.985 (initial) minus 0.01 per year
Radiation (TIMED-H)	.885
Radiation (TIMED-L)	.972

Figures 5-16 and 5-17 summarize available solar array power versus the observatory load requirement plus battery recharging. The observatory load power represents the power for subsystems and all instruments operated at a 25 percent instrument power duty cycle. A 25 percent power margin is added to the total observatory load estimate. Note that the available array power exceeds observatory load requirements throughout the mission life for both observatories.

Figures 5-18 and 5-19 illustrate the available instrument power duty cycle versus days from launch. These figures assume a spacecraft power load of 146 watts and 126.4 watts for TIMED-H and TIMED-L, respectively.

Significant additional instrument duty cycle is available during the early mission period. These power subsystem capability predictions account for the annual variation in solar intensity and assume the beta angle variation shown in Figure 5-14 and 5-15.

The instrument sprint mode consists of continuous operation for one orbit followed by three orbits to fully recharge the batteries resulting in a maximum battery discharge of 24.4 percent for TIMED-H and 27.8 percent for TIMED-L.

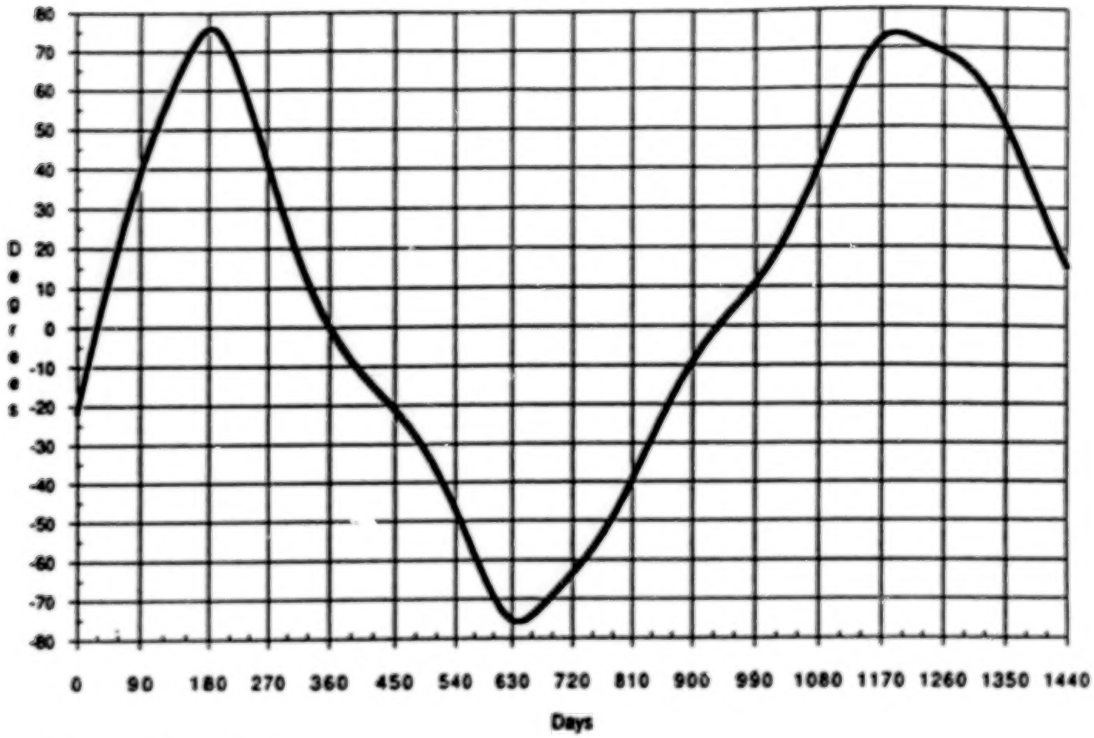
5.3.4 Reliability, Heritage and Risk Assessment

The survivability factor for the power subsystem is estimated at 0.992. Preliminary failure analysis is shown in Table 5-2. A single failure has no significant impact on the mission; however, power capability is reduced after one failure.

The body-mounted solar array consists of at least 18 strings combined in six shunt segments. Loss of a single string reduces the power capability by approximately 6 percent and results in reduced instrument duty cycle and/or reduced spacecraft subsystem operations.

The use of two batteries provides partial redundancy. Loss of either battery would require reduction of instrument duty cycle to limit the depth-of-discharge or would result in a reduced lifetime because of an increase in the depth-of-discharge. Limiting battery temperatures between 0 degrees C and 25 degrees C is critical to battery reliability.

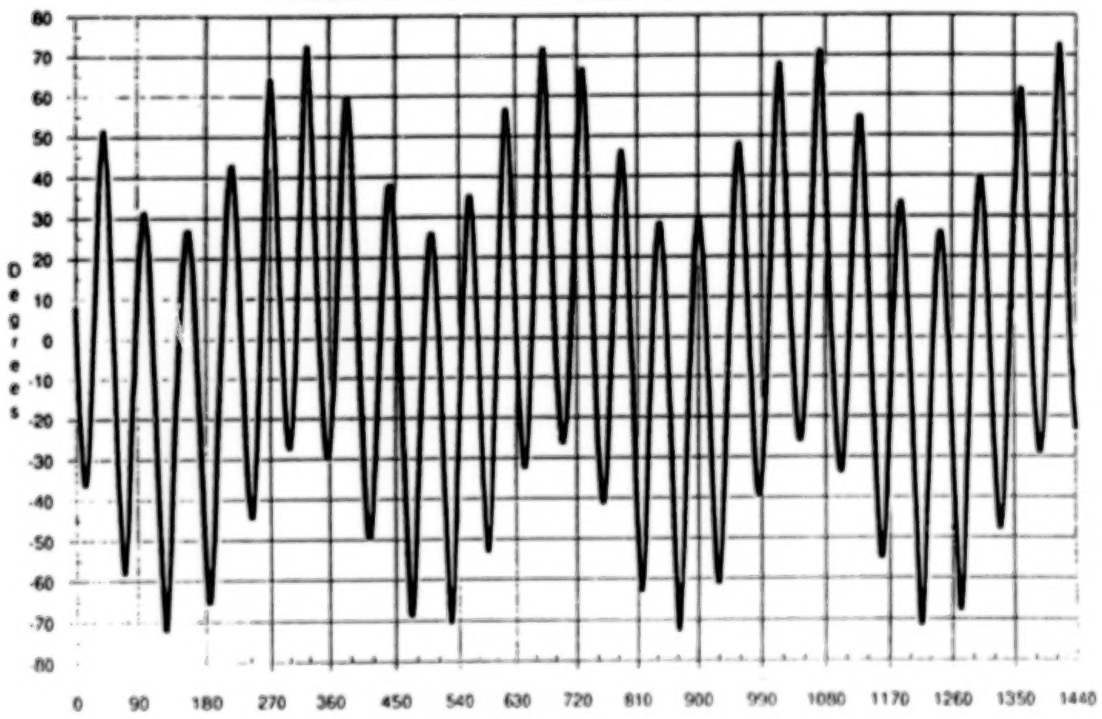
TIMED-H BETA ANGLE HISTORY



Launch Date = 2/1/1999 09:15 UT

Figure 5-14

TIMED-L BETA ANGLE HISTORY



Launch Date = 2/1999

Figure 5-15

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TIMED-H AVAILABLE AND REQUIRED POWER PROFILES

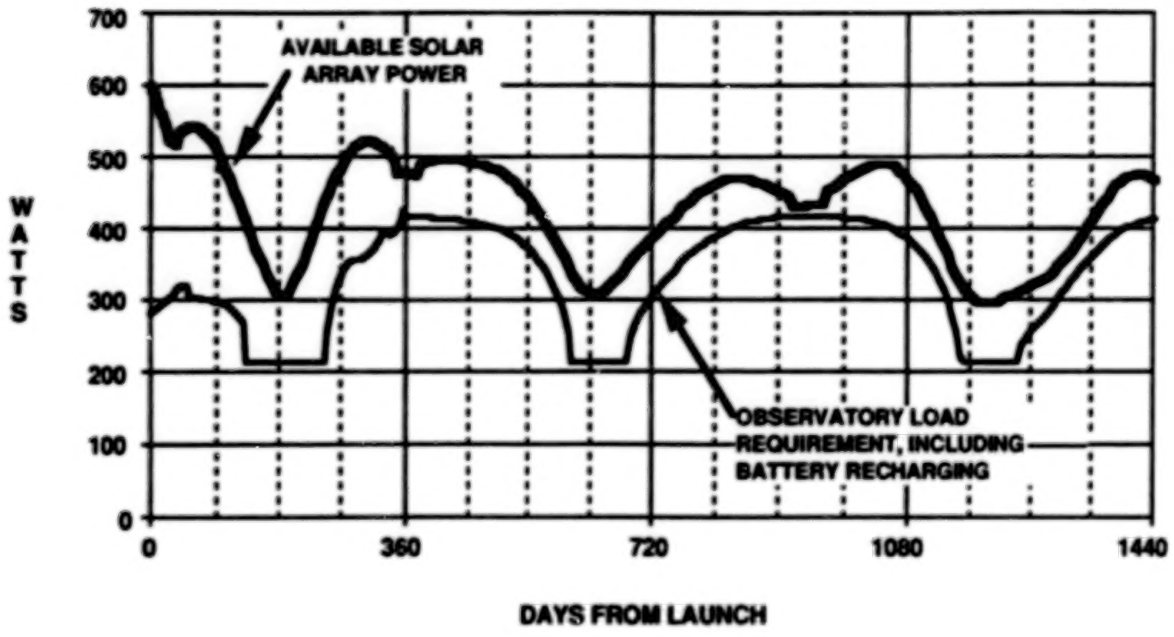

Figure 5-16

TIMED-L AVAILABLE AND REQUIRED POWER PROFILES

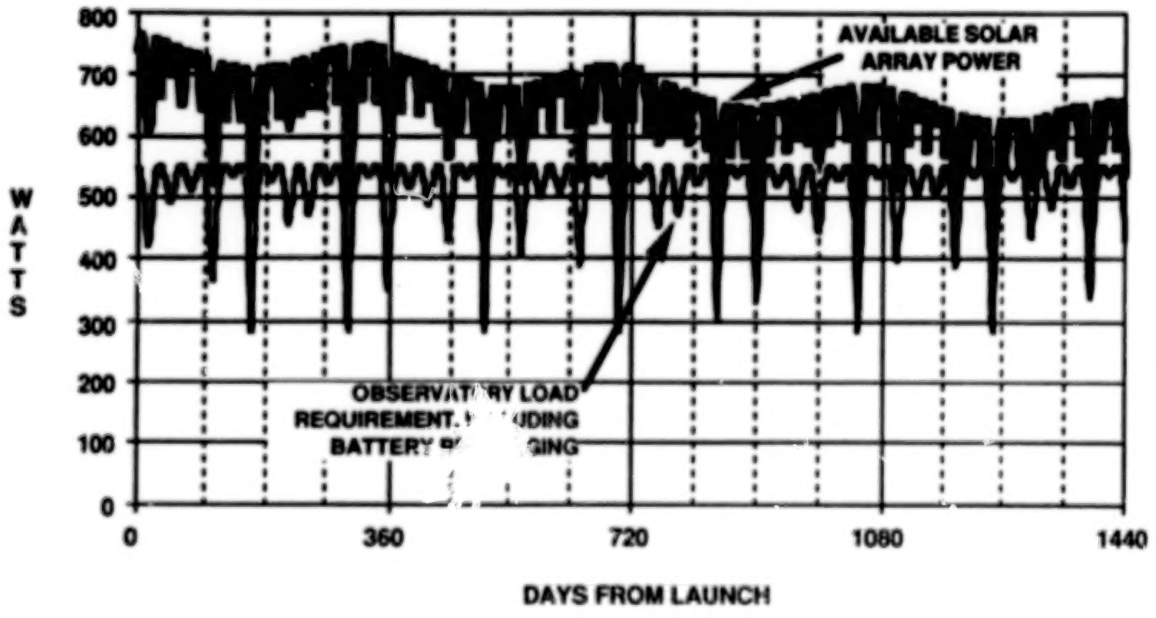

Figure 5-17

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TIMED-H INSTRUMENT POWER DUTY CYCLE CAPABILITY VS. DAYS FROM LAUNCH

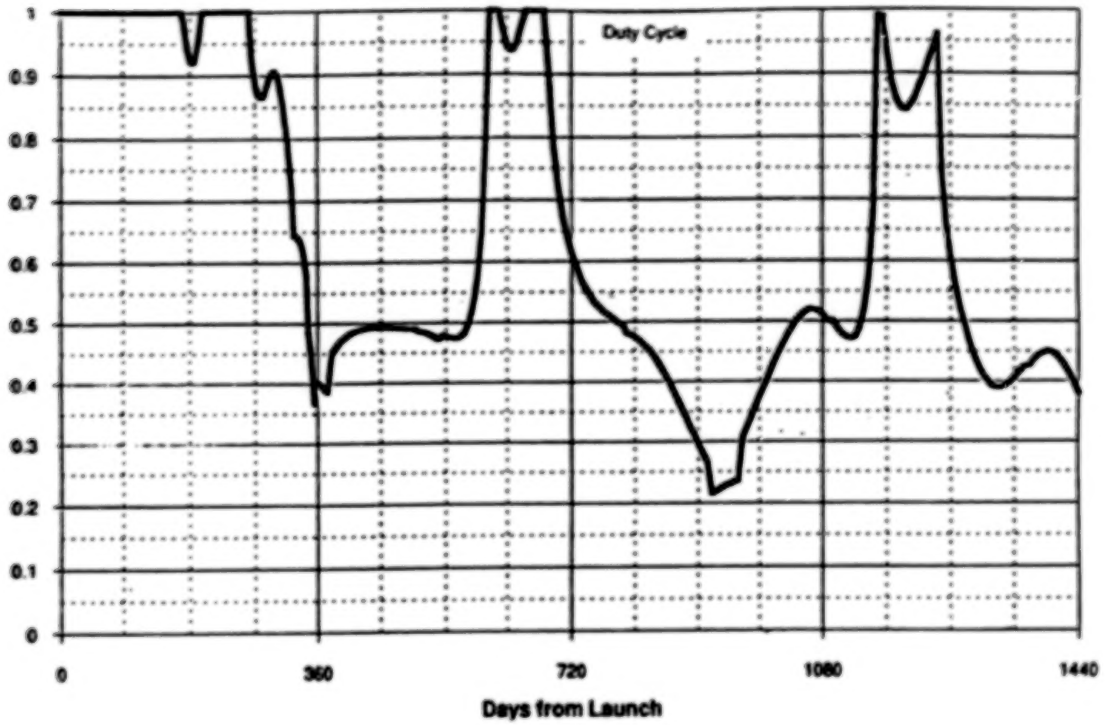

Figure 5-18

TIMED-L INSTRUMENT POWER DUTY CYCLE CAPABILITY VS. DAYS FROM LAUNCH

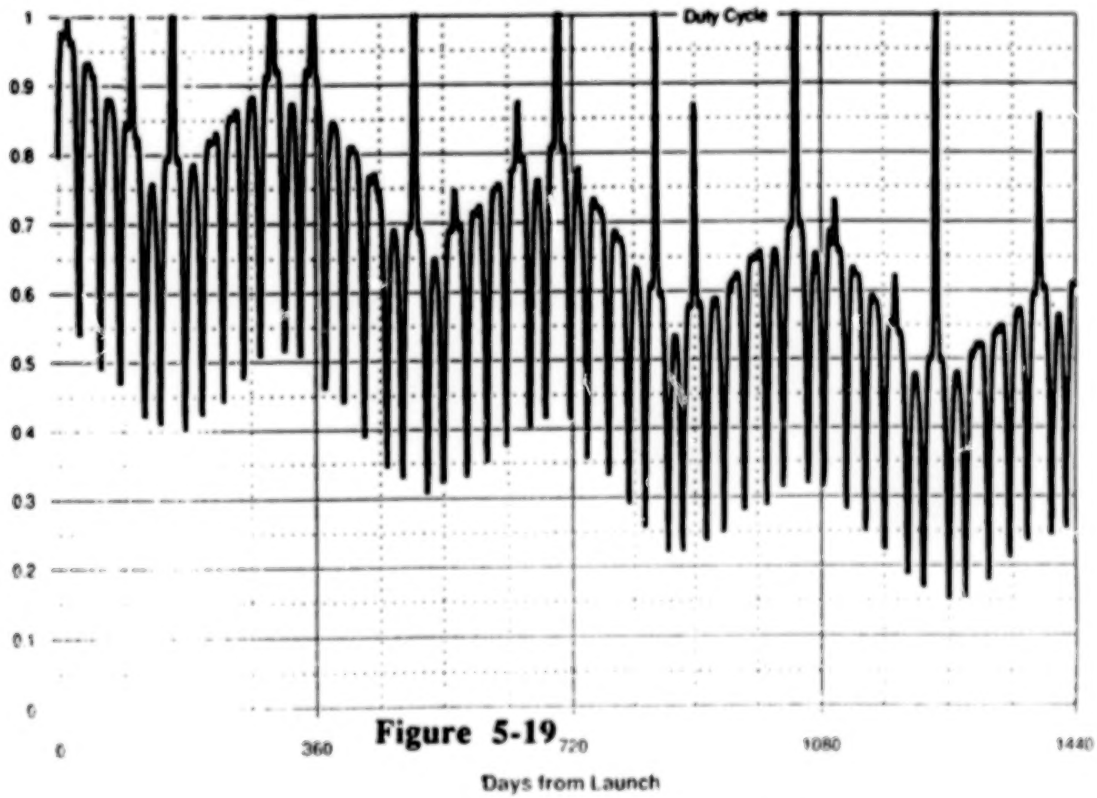

Figure 5-19

TIMED POWER SUBSYSTEM FAILURE EFFECTS ANALYSIS

Failure	Corrective Action	Mission Impact
Solar Array String-Open.	Isolated parallel strings provide power.	Loss of approximately 6 percent of generated power. Reduced duty cycle capability.
Solar Array String-Short.	Isolation diode reverse bias.	Loss of approximately 6 percent of generated power. Reduced duty cycle capability.
Battery Cell-Short.	Operate at reduced capacity.	Reduced duty cycle or mission life.
Battery Cell-Open.	Use single battery.	Reduced duty cycle or mission life.
PSE Charge Controller Fails.	Operate other charge controller.	Reduced duty cycle or mission life.
PSE Shunt Controller.	Use redundant circuitry.	None.
Shunt Dissipator Element Fails Open.	Use remaining shunts operating at increased current level.	None.
Shunt Dissipator Element Fails-Short.	Isolated by relay or fuse.	None.
EED Power Unit Relay Fails-Open.	Use redundant circuit.	None.
EED Power Unit Relay Fails-Short.	Two relays in series require separate arm/safe commands.	None.

Table 5-2

The power supply electronics incorporates redundancy in all critical circuitry, such as the shunt controller while each battery charger is zero fault tolerant. The loss of a single shunt dissipator element results in either a parasitic load on the bus on inability to shunt all the excess solar array power.

The direct energy transfer (DET) design has significant heritage with the Explorer spacecraft series, beginning with IMP, and was recently used on the COBE spacecraft. The array design and components are straightforward. Power subsystem component heritage is tabulated in Table 5-3. Gallium-arsenide solar cells have been flown on several classified Air Force spacecraft, and recently on the GSFC SAMPEX spacecraft. They are planned for the TRMM spacecraft.

The high performance super NiCd batteries have been flown on the GSFC SAMPEX mission in low Earth orbit and this technology should be well demonstrated prior to starting the TIMED detailed design.

TIMED POWER SUBSYSTEM COMPONENT HERITAGE

COMPONENT	HERITAGE
GaAs/Ge Solar Arrays	Classified Air Force Projects. SAMPEX and TRMM (Results available prior to TIMED detailed design.)
High Performance Super NiCd Batteries	SAMPEX (Low Earth orbit results available prior to TIMED detailed design.) Alternate use of larger standard NiCd batteries with extensive heritage can be substituted, if necessary.
Sequential Shunts	IUE, ISEE, COBE and SMEX

Table 5-3

5.4 Attitude Control Subsystem

5.4.1 Requirements

The attitude control subsystem (ACS) satisfies the following requirements:

- Provide observatory control during normal operation (1 revolution per orbit) and up to four revolutions per minute for brief periods to permit instrument calibration.

- Provide observatory pointing control within 1.0 degrees (3 sigma) in all three axes except during deep perigee passes on TIMED-H when an error of 4.0 degrees is acceptable in the roll axis only; i.e., roll is around the velocity vector. The roll error shall be removed prior to the next deep perigee pass.

- Provide attitude knowledge to an accuracy of 0.3 degrees (3 sigma) in all three axes, when measured at the instrument boresight.

- Provide control of center-of-gravity/center-of-pressure (cg/cp) offset to 1 centimeter.

- Provide observatory control during orbit adjust and 180 degrees yaw turn maneuvers.

5.4.2 Subsystem Description

The ACS is a conventional momentum bias system, shown in Figure 5-20, that provides observatory control about the pitch axis in both the one revolutions per orbit (rpo) and the four revolutions per minute (rpm) modes. The pitch axis is controlled by varying the speed of the momentum wheels which also provide control stability in the roll and yaw axes. A triad of magnetic torque rods provide momentum unloading capability and are the primary method for removing the disturbance created by the observatory during deep perigee passes. Attitude information is provided by a digital sun sensor and two Earth sensors. A set of center-of-gravity compensators are employed on TIMED-H only to provide on-orbit mass balancing.

The attitude information from the magnetometer, sun sensor, and two Earth sensors is routed to the attitude control electronics (ACE). The electronic signals from the sensors are processed by the interface electronics in the ACE so that they are compatible with the bus interface unit (BIU) and the safe mode controller. The sensor signals are then sent to the communications and data handling (C&DH) subsystem where the signals are included in the telemetry sent to the ground and also to the control processor. The control processor contains the control software that processes the input from the sensors and determines the attitude and rate of the observatory around its control axes. These control algorithms use this information to generate the required command signals for the momentum wheels and torque rods. During the acquisition mode, the control processor uses the rate and position information to damp the spacecraft rates and to acquire the mission attitude. The control algorithms supply the command for the torque bars and momentum wheels. During the mission mode, these control algorithms point the spacecraft nadir towards the Earth, as defined by the pointing requirements, and maintains this attitude against the disturbance torques. During the Delta V maneuver, the control processor generates the commands to the thrusters to maintain the thrust attitude relative to the commanded attitude and that measured by the Earth sensors and sun sensors. An off modulated control law is used to selectively turn off jets so that fuel is not wasted and observatory attitude control is maintained.

During the yaw turn, the control algorithms maintain the nadir pointing while the spacecraft is rotated in yaw. The torque rods provide the required damping and the yaw thrusters provide the yaw turn torque.

TIMED ATTITUDE CONTROL SUBSYSTEM BLOCK DIAGRAM

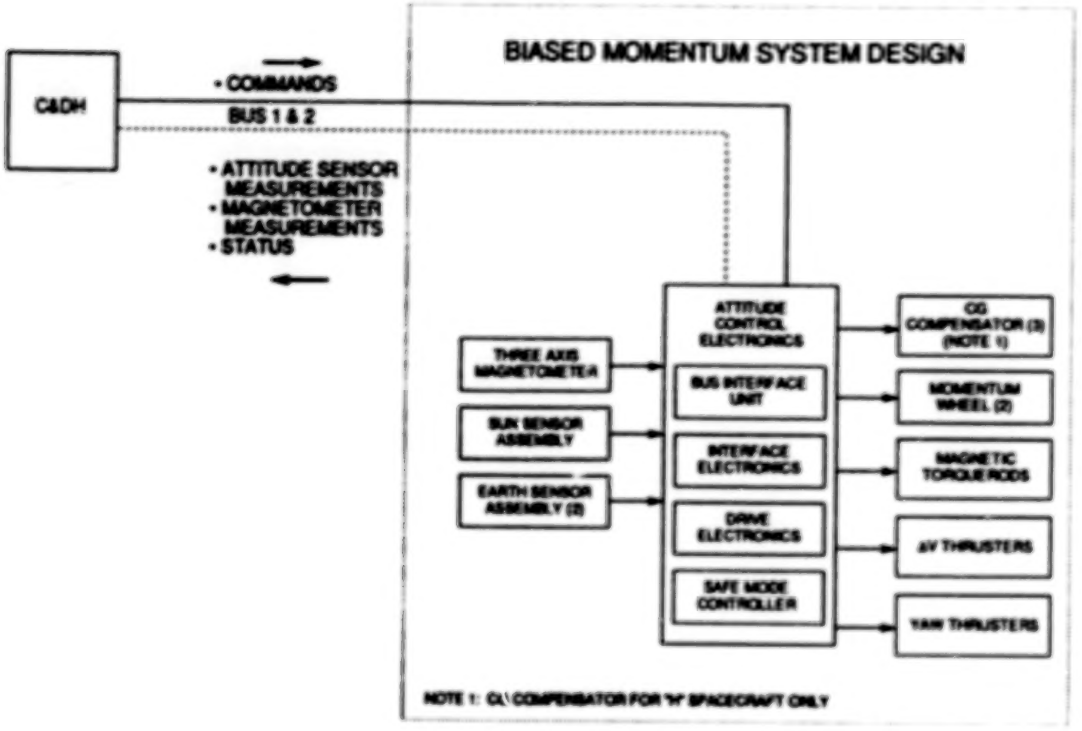

Figure 5-20

The yaw turn can be performed open loop if sufficient ground coverage is available, otherwise it can be performed by stored command that enables and disables the yaw jets. The sun sensor would be the controlling sensor for this maneuver. The control signals for firing the thruster are generated in the control algorithms of the control processor and fed back to the ACE for command of the thruster drivers. The torque rods' commands are fed back to the ACE via the 1773 data bus and the BIU in the ACE. From the BIU, the torque rods' signals are fed to the interface electronics and then to the drive electronics. The drive electronics are connected to their respective momentum wheels and magnetic torque rods.

If the control processor is inoperative due to failure, then the sensor signals are fed from the interface electronics to the safe mode controller. The safe mode controller generates torque commands that are routed to the interface electronics and then to the drive electronics.

If ground commanding is required, these commands will pass from the C&DH subsystem to the BIU and then to the interface electronics.

Each major subsystem component is described below in more detail.

Momentum Wheels

Two, redundant, momentum wheels each possessing 81 newton meter second (NMS) provide the primary control authority. The Honeywell Model HR60 is the selected momentum wheel based on momentum capability, power consumption, availability, and weight. The drive motor windings are modified to minimize spacecraft power. The momentum wheels are mounted with the spin axis parallel to the observatory spin axis.

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Magnetic Torque Rods

The magnetic torque rods are sized to eliminate the excess momentum produced by the observatory during the low perigee passes within one orbit (e.g., start the next perigee pass with no error from the previous pass). The magnetic moment capability is increased to approximately twice this requirement to decrease the time these torque rods would be operating, thus minimizing the impact on instrument operations and to take advantage of the stronger magnetic field of the Earth at the lower orbital altitudes.

The Ithaco Model D43111 is the selected magnetic torque rod, and three are positioned in an orthogonal triad located to minimize the impact of the magnetic fields on the instrument operation and the observatory magnetometers. Each torque rod provides 500 ATM^2 of capability while operating in the linear power region. Operation of these torque rods is in a continuously "on" condition or in an "off" condition with no proportional control capability. Dual windings are used for redundancy.

Large area coils were investigated as alternatives to the torque rods to reduce the power requirements and magnetic field impact on the instruments. The results indicate that the large area coils are less expensive, but the net power required is greater than the torque rods.

The use of propulsion jets for unloading the momentum disturbance was also considered. The magnetic torque rods could then be downsized to control the disturbances in the orbits that did not go to the minimum perigee altitude. The results are inconclusive because the power requirements are approximately the same since propulsion heater power is required. The amount of propellant required is small. The final design implementation depends on whether the selected instrument complement is more sensitive to magnetic field or hydrazine contamination.

Three-axis Magnetometer

The three-axis magnetometer measures the magnitude and direction of the external magnetic field relative to the observatory control axes. Magnetometer data is used for backup attitude determination and for controlling momentum unloading with the torque rods. The data can also be used for damping spacecraft motion using the torque rods in a B-Dot control mode.

The magnetometer selected is a SAM-63 series built by the Schonstedt Instrument Company. The input range is selectable and is typically 1000 milligauss. The attitude determination accuracy is between 1 and 5 degrees depending on altitude.

Sun Sensor Assembly

The sun sensor assembly provides two-axis attitude data for the observatory relative to the sun line and for attitude determination and yaw control.

The sensor consists of a sensor head and a separate electronics box. The attitude accuracy is 0.17 degrees in each axis and the field of view is ± 64 degrees. The field of view can be improved in 64 degree increments by adding additional heads, with minimum cost and weight penalty. The sensor head is located on the Sun facing end of the spacecraft along the pitch axis. This location provides observatory yaw and roll attitude relative to the sun line.

Earth Sensor Assemblies

Two Earth sensor assemblies provide pitch and roll attitude information for attitude determination and control and this information is used to point the spacecraft nadir towards the Earth and to control the pitch attitude of the observatory.

The use of two Earth sensors provides roll attitude accuracy that is independent of altitude by differencing the Earth chord as seen by the two Earth sensors. This is especially important for the highly elliptical orbits flown by TIMED-H.

The selected Earth sensors are conical sensors built by Ithaco Space Systems which consists of a sensor head and an electronics box. The sensor head is aligned along the spacecraft pitch axis and contains infrared optics, a scanning motor, and an amplifier. The infrared signal is deflected 45 degrees by a rotating prism and focused onto a detector. The width of the Earth pulse, and the relationship of the center of the pulse to the bottom dead center of the sensor head, are used to generate the roll and pitch errors. The electronics box can be located separately and contains electronics for power, command and telemetry. The second scanner head is located along the pitch axis on the opposite side of the spacecraft. The pitch error detected by the Earth sensors is used to control the speed of the momentum wheels and thus control the pitch attitude of the spacecraft.

ACS Electronics

The ACE contains the bus interface unit (BIU), interface electronics, drive electronics, and safe mode controller. The BIU provides the interface between the ACE and the C&DH subsystem. The interface electronics provide the signal conditioning and processing electronics necessary to interface with the BIU, drive electronics, and safe mode controller. The drive electronics contain the circuitry required to drive the momentum wheels and magnetic torque rods.

The safe mode controller contains the electronics necessary to put the spacecraft in a power and thermally safe attitude in case there is a failure in the C&DH on-board control processor. The safe mode controller would take command of controlling the observatory after it receives data that a failure has occurred. Using data from the magnetometers, Earth sensors, and sun sensors and the momentum wheels' and torque rods' commands as outputs, the ACE would control the observatory attitude by sending commands to the wheels and torque rods.

Center-of-Gravity (cg) Compensators

The TIMED-H observatory requires precise control of the cg/cp offset to minimize the aerodynamic torques. The propulsion subsystem provides some control over the cg location by selectively using propellant from each tank. However, a more precise control is required and this is provided by a triad of cg compensators. These mechanisms consist of a moveable mass on a threaded rod. The mass is moved along the length of the rod by a stepper motor.

The mechanism concept is adopted from the Global Geospace Science (GGS) Program, where they are used for dynamic balancing. This mechanism provides an inertia change capability of 1.4 $kg \cdot m^2$. This capability could be increased by using larger moveable mass and by increasing the 1 m length. This mechanism linearly translates a 2.7 kg weight over a distance of one meter. This is accomplished by driving a lead screw with a stepper motor/gear head, which precisely positions and holds the weight. The mechanism has end-of-travel stops to prevent over travel.

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5.4.3 Performance and Analysis

A parametric study was performed early to determine the performance of the biased momentum control system, identify critical parameters, define momentum wheel size and magnetic torque rod requirements, and quantify the pointing capability.

The major disturbance torque affecting the spacecraft is the aerodynamic disturbance torque which is a function of the air density and the displacement of the center of gravity from the center of pressure. The study approach was to determine how much disturbance was induced in the observatory around an orbit due to this torque, and to determine the sensitivity of this disturbance to cg/cp offset and air density (altitude of orbit perigee). The study also examined the effect of these disturbances on the pointing accuracy and rates in the roll and yaw axes.

Working with the TIMED Science Definition Team (SDT), it was established that the ACS performance required to go to a density equivalent of 120 km altitude was not feasible within the cost goals of the program. Figure 5-20 illustrates this point. A momentum wheel size of up to 1600 NMS would be required at the equivalent density of 120 km for small allowable roll angle disturbances. Based on this analysis and with SDT concurrence, atmospheric density equivalent to 130 km perigee altitude was baselined as the minimum perigee altitude.

Figure 5-21 also illustrates the ACS momentum wheel size sensitivity to the allowable roll angle error. The aerodynamic torque disturbance is manifested primarily as a roll error due to the gyroscopic effect. It was determined that a 4.0 degree roll error was acceptable for the defined instrument complement.

TIMED-H MOMENTUM WHEEL SIZE DESIGN DRIVERS

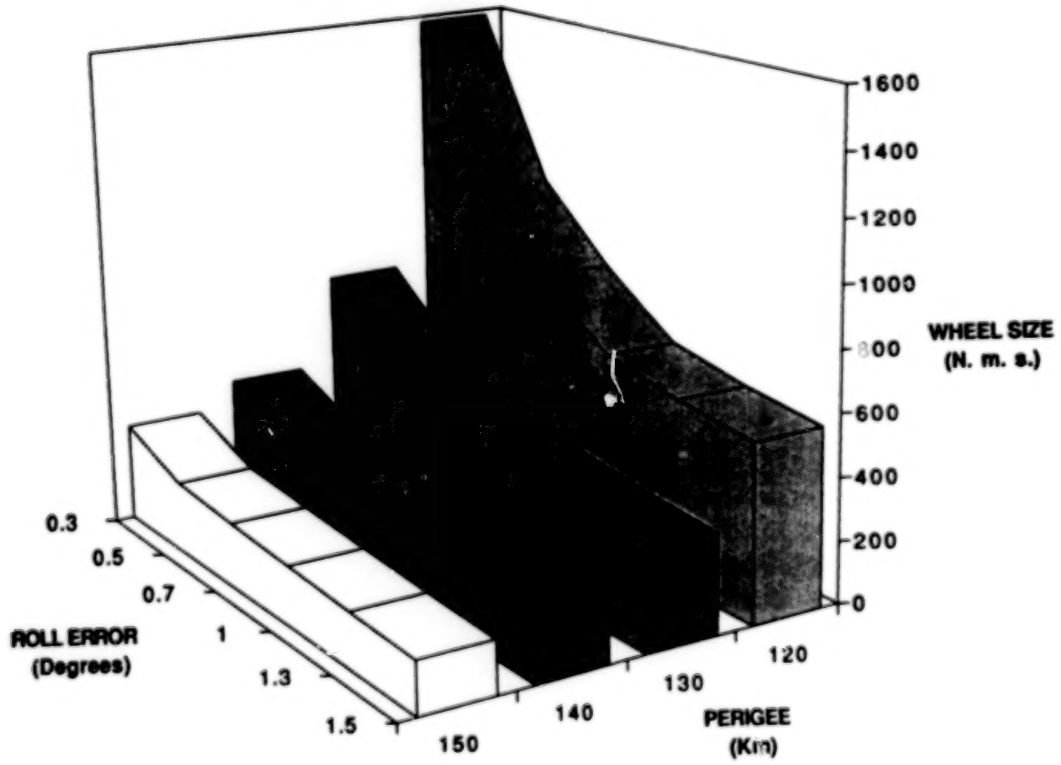

Figure 5-21

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The ACS performance for a 132 x 1000 km orbit with 81 NMS momentum wheels is shown in Figure 5-22. The figure shows the performance for two orbits of the spacecraft roll, pitch, and yaw angles. The performance is within requirements with the maximum roll angle at 4 degrees, the maximum yaw angle at -0.027 degrees, and the maximum pitch angle at 0.8 degrees.

There are two major observatory maneuvers supported by the ACS. The yaw turn maneuver, which is required to maintain the observatory cold end radiator facing the anti-sun direction, is accomplished using the zero momentum, open loop method. Initially, the momentum wheel is spun down, and using orbit adjust jets, observatory momentum is removed as Earth sensors provide observatory orientation. The yaw jets are fired to perform the actual maneuver. Using the orbit adjust jets, momentum is put back into the observatory, and since the pitch loop is still active, this results in momentum being transferred back into the momentum wheels.

Preliminary analysis indicates sufficient fuel and tank capacity to accomplish the required number of maneuvers. Time to accomplish each maneuver is approximately 3 orbits.

The orbit adjust maneuvers are necessary to maintain orbit altitude and accommodate the orbit profile and are accomplished by closed loop jet firings. Orbit adjust jets are fired around apogee and perigee. When the roll or pitch deadband is exceeded, the appropriate jet is off-pulsed. Pitch control is implemented by closing the pitch loop through the jets using Earth sensor data. In a similar manner, yaw control is established using yaw jets. However, small thruster offsets cause yaw torque that results in drift which will limit the burn duration. Experience from early burns will provide knowledge of the thruster offset and will permit adjustment of future burn schedules.

Preliminary analysis indicates that these maneuvers are feasible. The time it takes to accomplish the maximum Delta-V burns (approximately 70 M/S) is 2 to 3 days.

5.4.4 Reliability, Heritage and Risk Assessment

The survivability factor for the ACS is estimated at 0.9893 after 2 years based on a comparison with EOS and TOMS spacecraft reliability studies.

Selected ACS components are items currently in use, being designed, or have been previously used on spacecraft. A heritage listing is shown in Table 5-4.

The ACS is highly reliable due to the biased momentum system architecture, the heritage components selected and the use of selective redundancy. This redundancy is based on a preliminary evaluation of ACS component performance degradation and their impact on mission success. The effects of component failures on the ACS functions are shown in Table 5-5.

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TIMED-H ATTITUDE CONTROL SUBSYSTEM PERFORMANCE

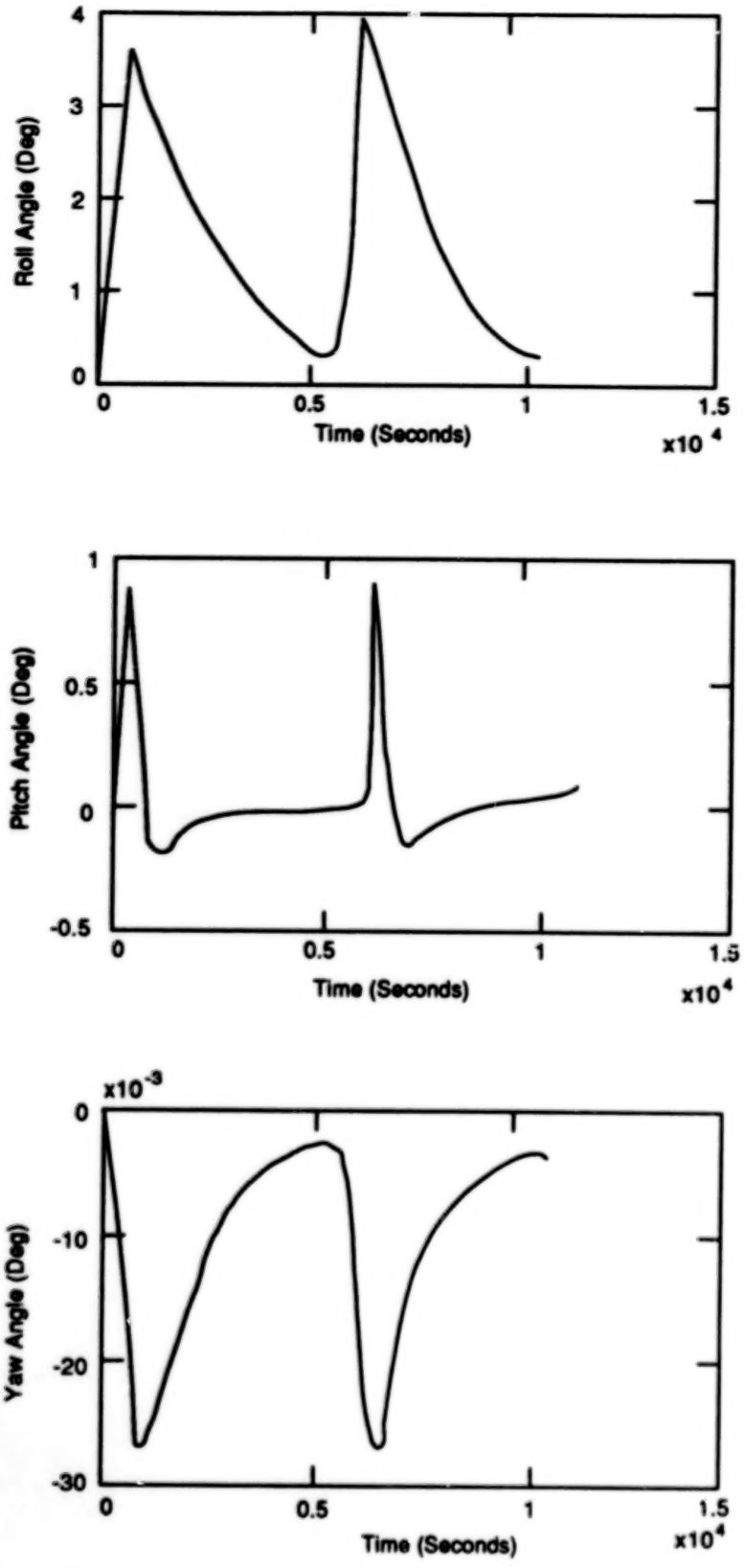

Figure 5-22

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TIMED ACS SUBSYSTEM COMPONENT HERITAGE

COMPONENT	HERITAGE
Earth Sensor Assembly	MMS MACS: LANDSAT,
	UARS, DMSP
Digital Sun Sensor Assembly	MMS MACS: SMM, LANDSAT, UARS
3-axis Magnetometer	MMS MACS: SMM
Magnetic Torque rods	UARS, GRO
Momentum Wheel	DSP
cg Compensator	GGS
Attitude Control Electronics	TBS

Table 5-4

TIMED ACS FAILURE EFFECTS ANALYSIS

Failure	Backup Mode	Mission Impact
c.g. Compensator (TIMED-H spacecraft only)	Use propulsion unloading. Increase torque rod duty cycle.	Degraded performance if momentum cannot be unloaded. Need to avoid lower orbits.
Bus Interface Unit*	Use redundant BIU.	No mission impact.
Interface Electronics*	Use redundant interface electronics.	No mission impact.
Drive Electronics*	Use redundant driver for momentum wheel and torque rods.	Same as momentum wheel failure or torque rod failure.
Safe Mode Controller*	Use redundant safe mode controller.	No impact.
Magnetometer	Attitude determination by Sun sensor and Earth sensors. Wheel unloading by ground command.	Attitude determination during eclipse. No yaw information. Momentum unloading more complex.
Sun Sensor	Use magnetometer for yaw attitude Earth sensors for roll attitude determination.	Some degradation in yaw and roll attitude determination. Sun angle data is lost.
Earth Sensor	Use redundant Earth sensor.	Degraded roll attitude information in highly elliptical orbit.
Momentum Wheel*	Use redundant momentum wheel.	Minimal impact because wheels are operated at 50 percent of capacity.
Torque Rod Coil*	Use redundant coil.	Limit maximum torque of torque rod.

*Redundant Units

Table 5-5

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5.5 Propulsion Subsystem

5.5.1 Requirements

The propulsion subsystem satisfies the following requirements:

- Provide observatory orbit adjust maneuvers and 180 degree yaw turn maneuvers.
- Provide capability to maintain the orbital altitude profile requirements derived from Figure 2-2 and depicted in Figures 5-23 and 5-24.
- Provide safe disposal of the observatory at the end of the mission or useful life.
- Restore orbit altitude to offset the effect of aerodynamic drag.
- Assist in controlling the location of the observatory cg.
- Minimize thruster center-of-thrust vector offset with respect to the cg so as to preclude excessive attitude disturbance torques.

5.5.2 Subsystem Description

The propulsion subsystem is a monopropellant, hydrazine, blowdown system with six, stainless-steel, spherical tanks. The system operates at a blowdown ratio of 4:1 with an initial pressure of 2.76 megaspascals. Total hydrazine load capacity at this blowdown ratio is 232 kg. Figure 5-25 shows the subsystem block diagram.

Each major subsystem component is described below in more detail.

Propellant Tanks

Fuel is equally contained in six 48 cm diameter tanks. A ring-reinforced, metallic diaphragm within the tank separates the fuel from the pressurant gas. The diaphragm is designed to progressively collapse and follow the contour of the propellant as it is depleted over the mission life. This controlled diaphragm operation provides positive location of the propellant during mission life for cg control considerations.

Each tank may be fueled with 36.32 kg of hydrazine at the design beginning of life (BOL) pressure of 2.76 MegaPascals. Tank expulsion efficiency is 98.5 percent, resulting in useable fuel of 35.78 kg per tank.

ARDE Model E34407 is the selected baseline tank and is currently in production at ARDE.

Thrusters

Catalytic decomposition, hydrazine thrusters provide a range of thrust (blowdown pressure dependent) for orbital maneuvers.

Four Rocket Research, Model MR-50S 22.2 N, thrusters are selected for orbit adjust ΔV maneuvers. The selection is primarily due to the large propellant throughput required to satisfy the TIMED-H orbital profile. The thrust magnitude is a compromise to minimize ACS control authority problems and to provide adequate acceleration levels for performing

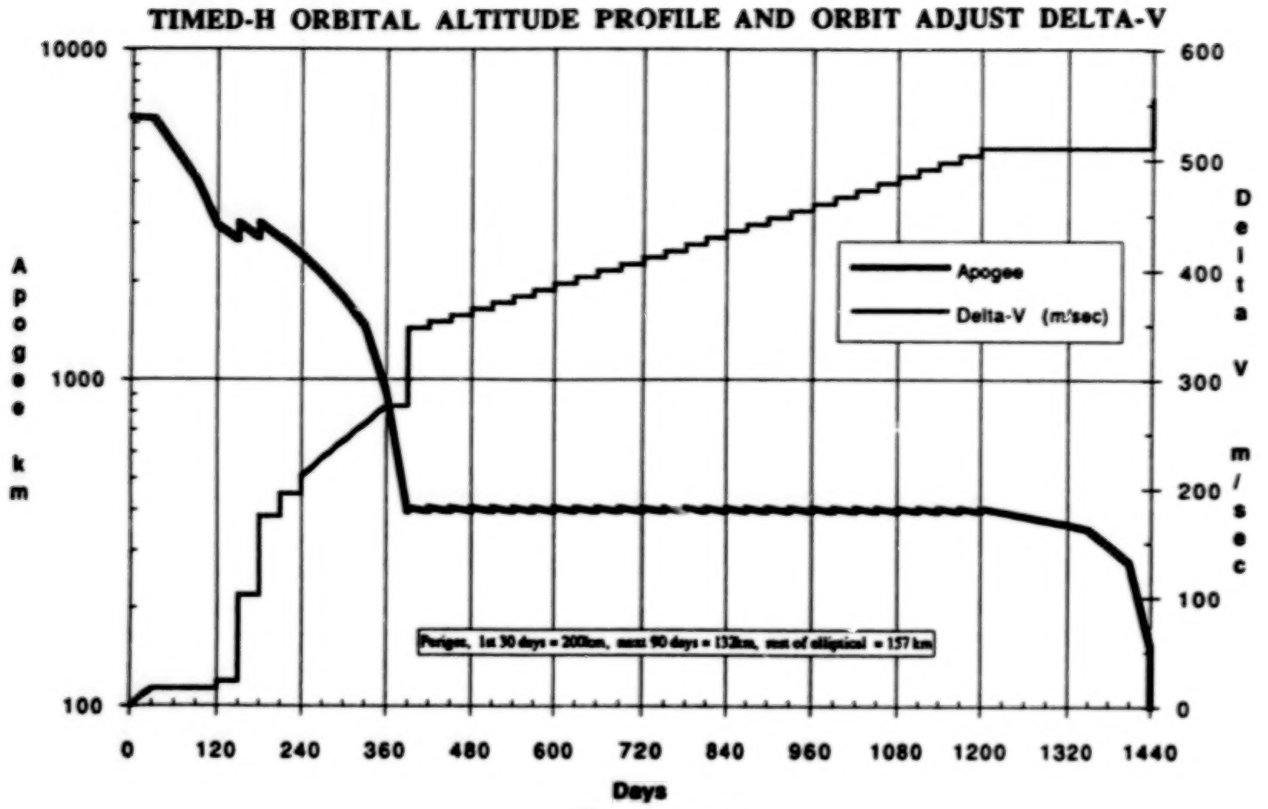

Figure 5-23

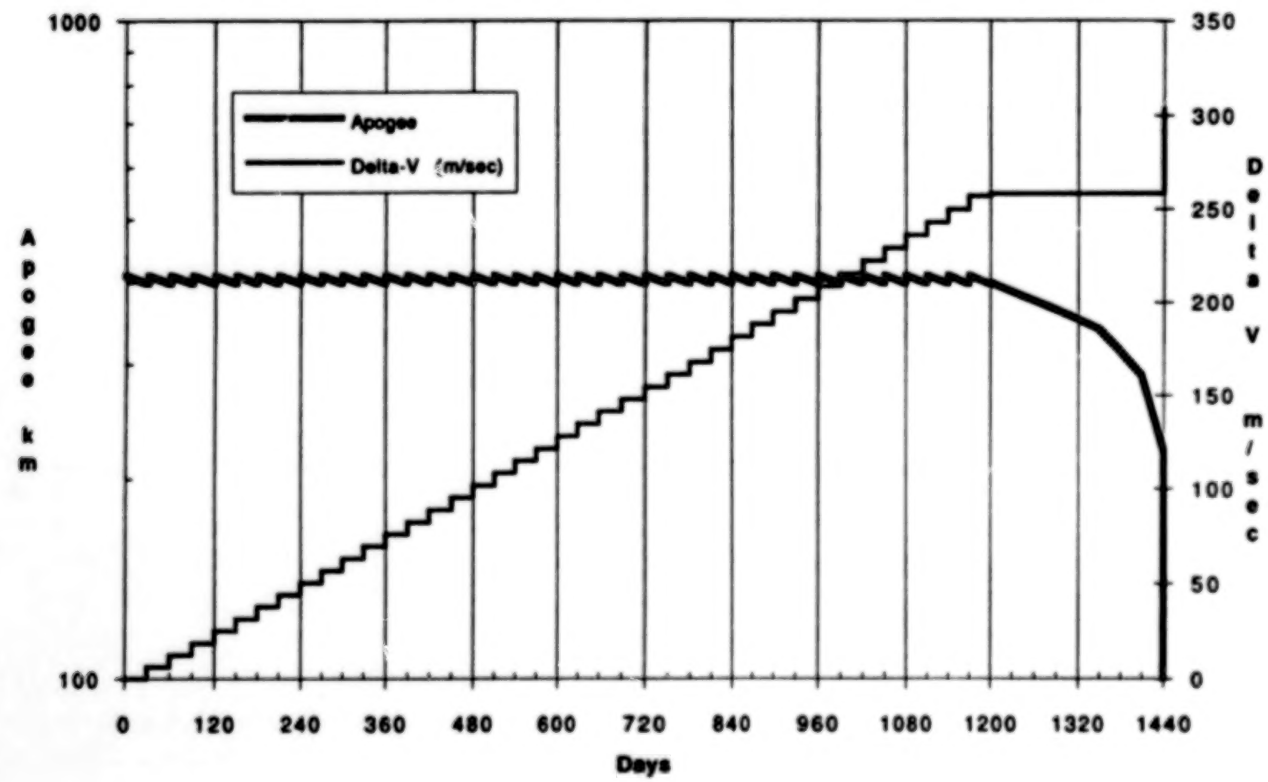

Figure 5-24

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PROPULSION SUBSYSTEM BLOCK DIAGRAM

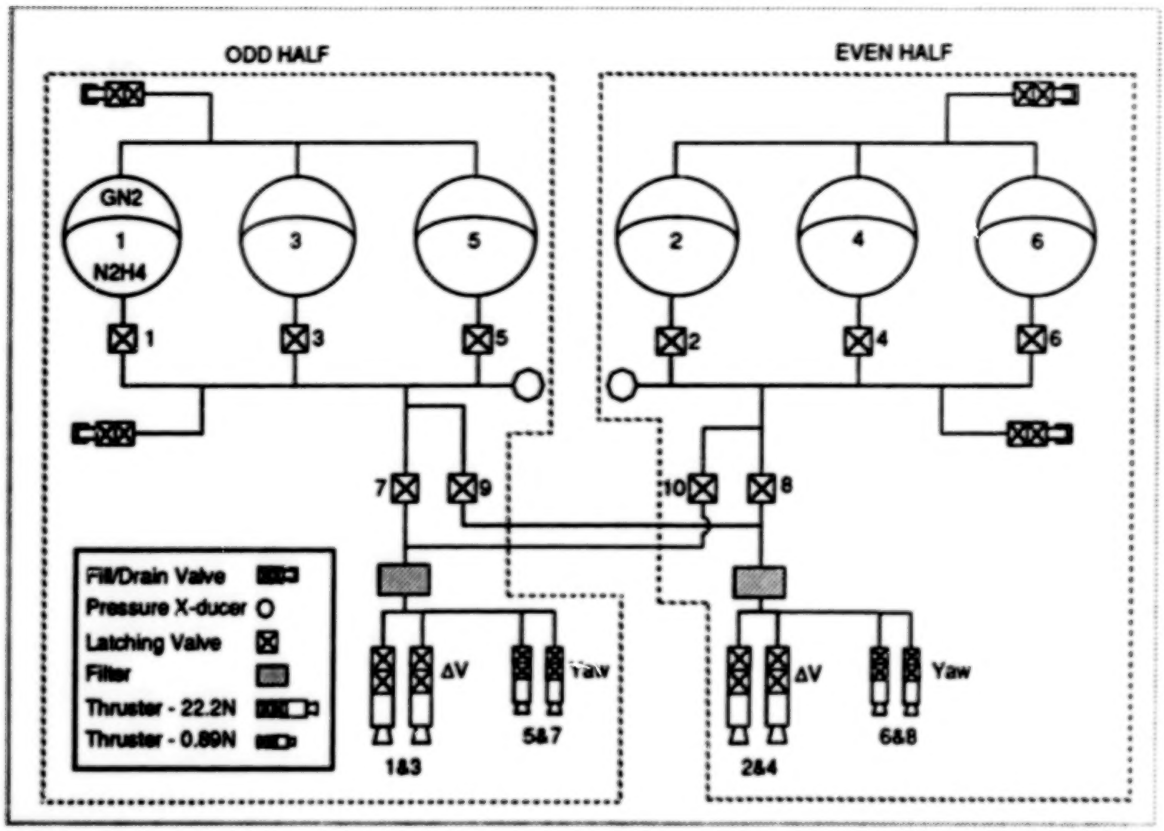

Figure 5-25

the large ΔV maneuvers. For this application, thrust and specific impulse (Isp) will vary from 32.2 to 22 N and 227.5 to 220 sec respectively, depending on tank blowdown pressure.

Two 0.89 N thruster pairs, providing mutually opposed yaw torques, are provided to accomplish spacecraft yaw maneuvers. Rocket Research Model MR-103 is the selected baseline. Thrust and Isp vary from 1.07 to 0.68 N and 224 to 219.5 sec respectively, depending on tank blowdown pressure. Thruster physical geometry is shown in Figure 5-26.

All thrusters receive their firing commands from the ACS logic and circuitry. Catalyst bed preheating for approximately 30 minutes is part of the operational enabling sequence and consumes approximately 4 to 10 watts of power for this time depending on the thruster size. Actual firing is accomplished via application of power from the PSE to the thruster valve body dual solenoid drive coils.

Isolation Valves

Fuel is to be selectively supplied to system manifolds and thrusters from any tank by programmed or commanded opening of individual latching isolation valves at the outlet of each tank. This feature is required to provide a coarse observatory cg control capability.

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TIMED BASELINE THRUSTER GEOMETRY

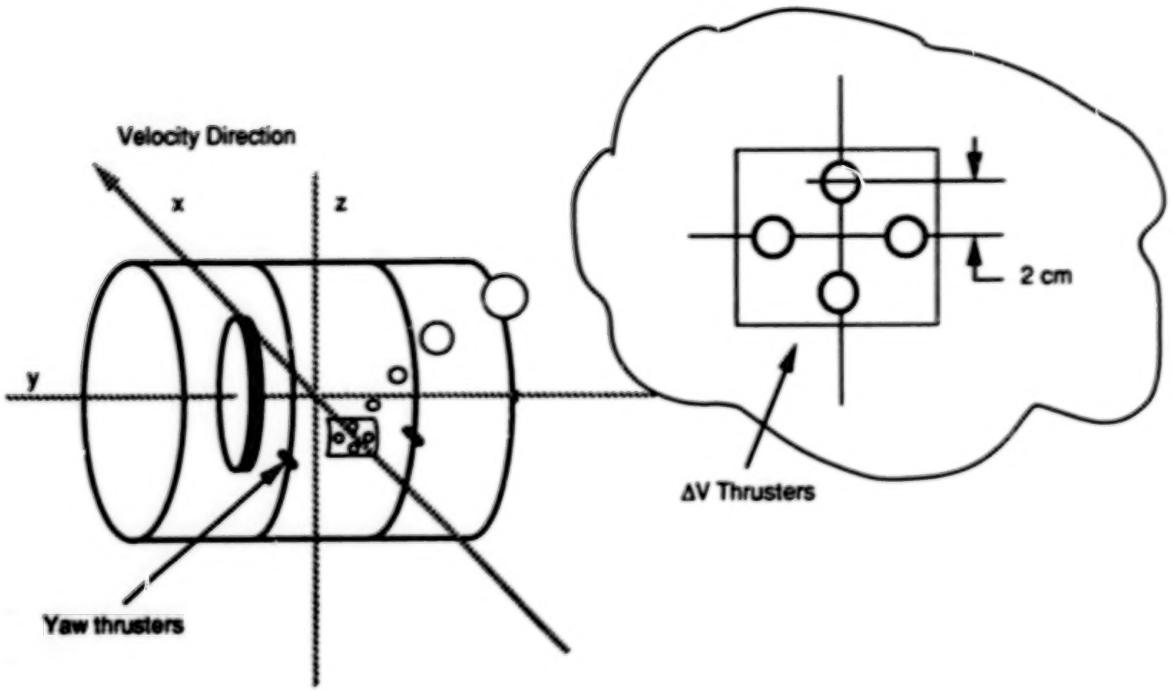

Figure 5-26

The tank outlet isolation also precludes shifting of propellant within the half-system by thermal pumping. Two latch valves isolate the redundant thruster banks, while two additional valves provide the cross-strapping capability.

MOOG, Inc., Model 51-106, is the selected baseline valve.

5.5.3 Performance and Analysis

The propulsion subsystem has a total impulse capability of 509,090 N-sec when the tanks are fueled to the design capacity. The required total impulse for the TIMED-H and TIMED-L spacecraft are 478,368 and 258,225 N-sec respectively.

The TIMED-H orbit profile requires a velocity change capability of 557 m/sec, including reserve for safe disposal, equating to a fuel expenditure of 218 kg. With a total fuel capacity of 232 kg, 14 kg of fuel remains for yaw maneuvering or margin.

The TIMED-L orbit profile requires a velocity change capability of 302 m/sec, equating to a fuel expenditure of 116 kg. Since total fuel capacity is 232 kg, 116 kg of reserve fuel is available for 180 degree yaw turn maneuvering or margin.

The expected orbital profiles are shown in Figures 5-23 and 5-24. The atmospheric density assumed for the minimum perigee and circular orbit mission phases are 6.0×10^{-9} and 1.0×10^{-11} kg/m^3, respectively.

Fuel requirements are calculated using observatory dry mass, ΔV required and thruster Isp. Since Isp does not vary linearly with tank blowdown pressure, a weighted average is used so that Isp is not underestimated.

Selective depletion of tanks for cg control is possible due to isolation valves at the outlet to each tank. An analysis was performed to determine the expected accuracy of propellant gauging once the observatory is in orbit. Using thermodynamic P-V-T relationships and accounting for telemetry and measurement errors, it was determined that propellant mass could be known within 2.2 percent of the actual fuel amount in the tank(s).

No fuel is budgeted specifically for performing the 180 degree yaw turn maneuvers, but the fuel margin above that required to supply ΔV is assumed to be adequate to also support the yaw maneuvers.

5.5.4 Reliability, Heritage and Risk Assessment

The survivability factor for the propulsion subsystem is estimated to be 0.996 after 2 years based on an assessment using component reliability data gathered from the NASA EOS and TDRS spacecraft reliability studies. Subsystem reliability is high due to design redundancy.

The subsystem design is considered very low risk since heritage components with flight histories are selected for the design. Component heritage is described in Table 5-6.

TIMED PROPULSION SUBSYSTEM COMPONENT HERITAGE

Component	Heritage
Tanks	Sandia National Labs, Classified
Thrusters	ISTP-GGS, GPS IIR
Latching Isolation Valve	Hughes HS376
Fill/Drain Valve	TIROS, DMSP, Landsat
Pressure Transducer	ISTP-GGS, Satcom, GPS IIR
Filter	TIROS

Table 5-6

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Preliminary failure effects analysis is described in Table 5-7.

TIMED PROPULSION SUBSYSTEM FAILURE EFFECTS ANALYSIS

Failure	Backup mode	Mission Impact
ΔV thruster fails	Use redundant thrusters	Degraded system performance, possible attitude control impact
Yaw thruster fails	Use redundant thrusters OR magnetic torque rods	Increased time to perform yaw maneuver
Tank isolation valve fails shut	Other tanks	Loss of fuel fraction. Reduced mission life. Possible cg control impact. Minimum altitude increased.
Tank isolation valve fails open	None	Possible cg control impact. Minimum altitude increased.
Fill/Drain Valve	Redundant seat w/cap	None
Pressure Transducer	Redundant backup	Possible reduced gauging accuracy
Filter	System isolation with redundant half-system	None
Thruster leak	System isolation with redundant half-system	None

Table 5-7

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5.6 Thermal Control Subsystem

5.6.1 Requirements

The thermal control subsystem (TCS) satisfies the following requirements:

- Maintains the temperature and gradients of all elements of the observatory within specified limits under all expected mission environments. These environments include qualification and acceptance testing, launch operations, ascent and deployment, orbit maneuvers, and orbit operations.

- Maintain body-mounted solar cells temperature so that reasonable power can be generated. Maximum solar cell temperatures of 60 degrees C for the cylinder mounted and 70 degrees C for the end mounted are the design requirements.

- Maintain all temperature limits when the solar beta angle (angle between the orbit plane and sun) varies between ±90 degrees.

- Accommodate the effect of aerodynamic heating during the low perigee portions of the orbit.

- Provide special heat sinks for the IRLS instrument to dissipate 50 watts at -5 degrees C, and for the NIRS instrument to dissipate 7.5 watts at -5 degrees C.

- Operate during observatory spin rate of 4 rpm.

5.6.2 Subsystem Description

The thermal design approach is to provide temperature control for the body-mounted solar cells while maintaining an internal observatory environment which is thermally isolated and independently controlled by the use of a louvered, anti-sun radiator. Thermal management of the body-mounted solar cells is provided by a matrix of heat pipes embedded in (or mounted to) the solar panels as shown in Figure 5-27. Multilayer insulation (MLI) is provided behind the solar-celled panels and, together with low conductivity standoffs, provide thermal isolation for the components and instruments inside the observatory. The heat pipes, which are attached to the solar panels, are thermally connected to provide heat sharing between the sun and anti-sun sides of the observatory, regardless of the solar incidence angle. This heat sharing reduces the temperature of the solar cells and greatly increases their power generation capability.

Most of the dissipated energy from the instruments and subsystem components is transferred by conduction and radiation to a dedicated cold end radiator which contains louvers and is maintained between 5 to 20 degrees C for all orbital environments. Six heat pipes are employed to transfer the instrument heat load to this radiator. The subsystem components are mounted to the in-board side of the radiator and have a good heat transfer path directly to space.

Based on a total radiator area equal to 2.5 m^2 on the anti-sun end, the TCS is capable of handling internal heat loads of up to 550 watts with 1.2 m^2 of louver area.

The TCS hardware items consist of heaters, thermostats, heat pipes, multilayer insulation, thermal coatings, and louvers. A brief description of each item is given below:

THERMAL CONTROL SUBSYSTEM CONCEPT

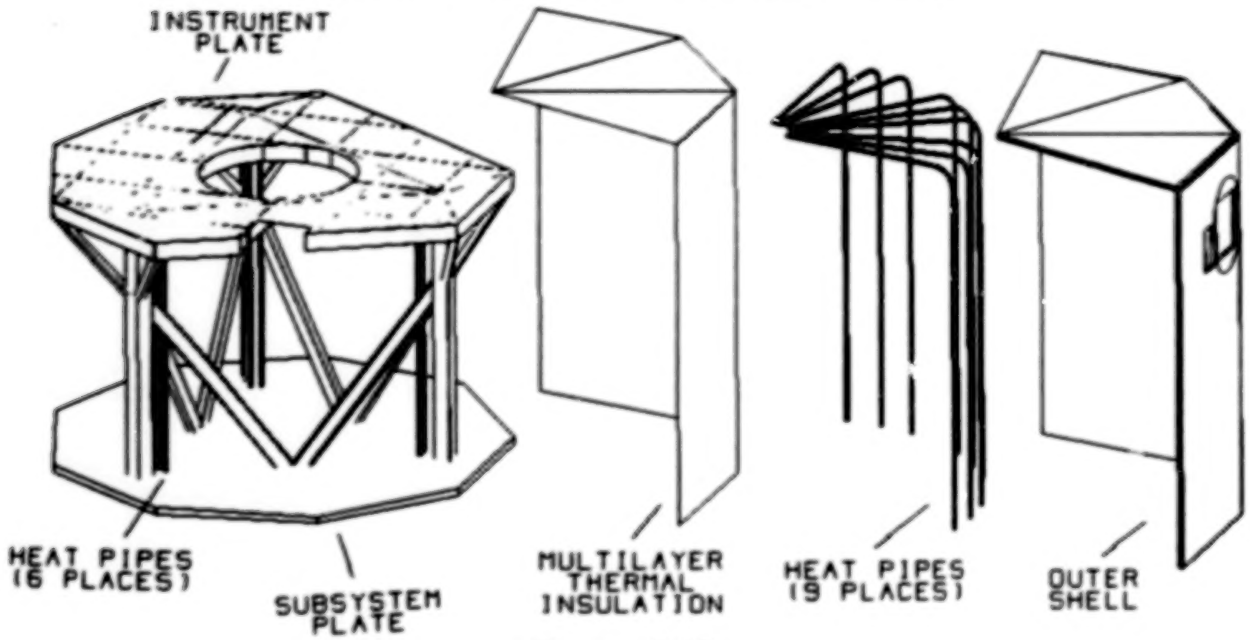

Figure 5-27

Heaters

The heaters are the thin-wire resistance type, Ninco Thermofoil or equivalent, with Kapton insulation. Although the design minimizes the need for heaters during the cold environment, some heaters with thermostatic control are required for the temperature sensitive hydrazine tanks and lines. These components have a minimum design temperature limit of 5 degrees C.

Heat Pipes

Each heat pipe is 1.25 cm in diameter, axially grooved, and use ammonia as the working substance. Twenty-seven L-shaped heat pipes are provided to distribute the absorbed solar energy which is converted into heat over the exterior surface of the spacecraft. The heat pipes are spaced on 23 cm centers around the circumference of the observatory. The limitations imposed by the length and heat transport capability (about 90 W/m), require that each loop be divided into three individual pipes so that 81 individual heat pipes are required to maintain the solar cells. These heat pipes are mechanically coupled at the solar cell end of the observatory. They are thermally isolated from the space radiator at the other end of the observatory. The absorbtivity to emissitivity ratio for solar cells is assumed to be 0.82/0.80.

Internal heat pipes are used to maintain a near uniform temperature on the instrument and subsystem mounting plates as well as to provide temperature control for certain components and instruments by transporting their waste heat directly to the space radiator. Four are required for each mounting plate.

Four heat pipes are required to transfer the instrument mounting plate heat to the radiator. In the case of the TIMED-L spacecraft, two additional heat pipes are required to transfer 55 watts from the Sterling cycle cooler which services the IRLS instrument.

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MLI

Multilayer insulation blankets are required to protect the components from the relatively hot and on occasion cold body-mounted solar cells. The MLI is attached to the inside surfaces of the solar panels. They cover the entire surface of the observatory, except for the thermal radiator and the instrument and subsystem cutouts. The propulsion tanks and associated hardware are also covered with thermal blankets to control heat leaks.

The effective emittance of MLI is 0.02 in the cold case and zero in the hot case.

Thermal Coatings

The radiator end of the observatory contains the primary heat sinks for the IRLS and NIRS instruments and serves as the observatory radiator for the other subsystem component and instrument waste heat. The exterior surface of the radiator (about 2.5 m^2) will have a stable thermal coating such as S-13 GLO white paint. All interior surfaces of the observatory and exterior surfaces of the components will be coated with a high-emittance black paint such as Chemglaze black to enhance radiation interchange.

The absorbtivity to emissitivity ratio for white paint is 0.3/0.85.

Thermal Louvers

Vane-type thermal louvers are used over a portion of the space radiator to aid in maintaining the minimum temperature level during the cold case when component dissipation is at a minimum. The louvers are mounted directly to the external surface of the radiator baseplate and provide a variable emittance between 0.1 and 0.7 when the baseplate temperature varies between 5 and 20 degrees C. The louvered area required is 1.2 m^2.

5.6.3 Performance and Analysis

Exterior Thermal Analysis

This section includes thermal analysis for the body-mounted solar panels, aerodynamic heating effects and the deployable solar array.

The incident flux values used for the thermal analysis are:

Hot Case	Cold Case
Solar: 1419 w/m^2	Solar: 1286 w/m^2
Earth: 265 w/m^2	Earth: 208 w/m^2
Albedo: 0.25	Albedo: 0.35

Figure 5-28 is a plot of the solar flux values for each of three orbital cases with beta equal to 0: a circular orbit at 400 km, and two highly eccentric orbits with apogees at 12000 and 6000 km and with perigee located at the south pole. This figure illustrates that the total flux and the thermals gradients increase with orbital altitude which requires that the TCS must distribute the absorbed energy over the entire surface of the observatory to maintain acceptable solar cell temperatures.

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ORBITAL AVERAGE INCIDENT SOLAR FLUX VARIATION WITH ORBITAL ATTITUDE

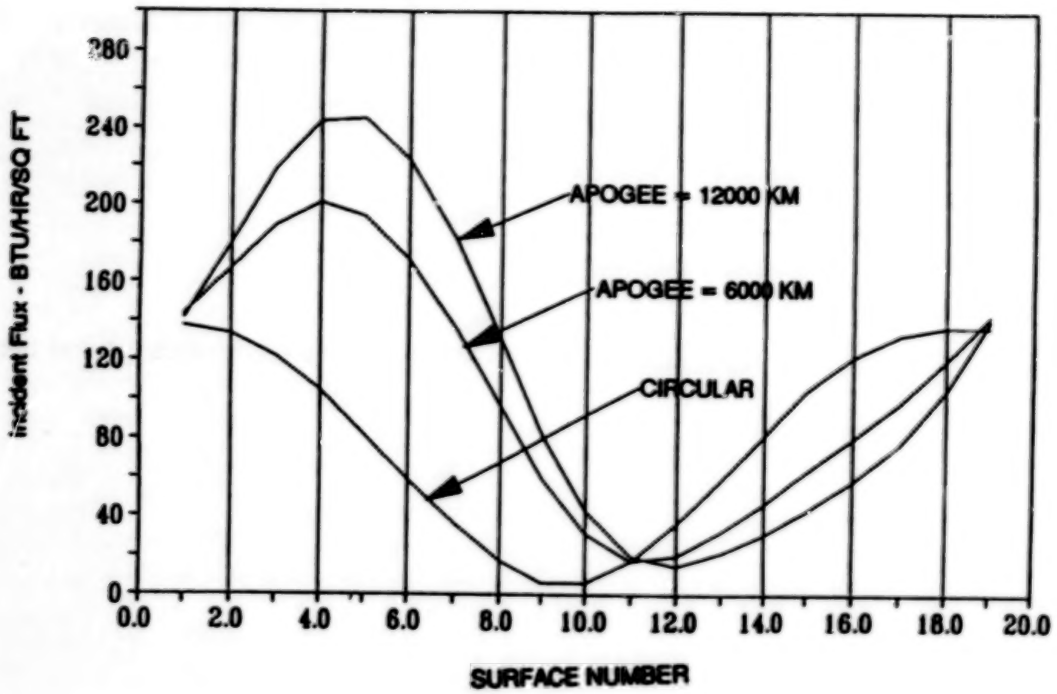

Figure 5-28

TIMED THERMAL ANALYSIS MODEL

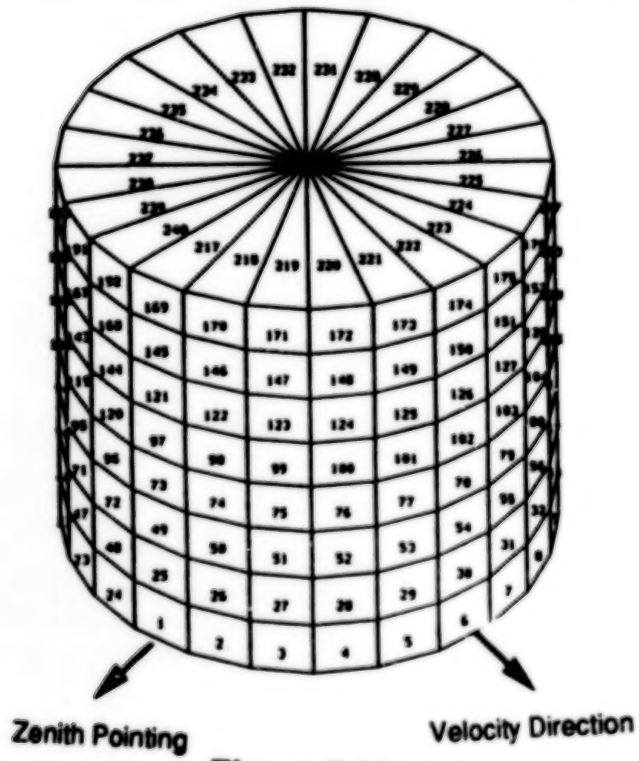

Figure 5-29

086

A multi-node transient model, in the SINDA format, was generated to evaluate the thermal behavior of the body-mounted solar cells in various orbits as illustrated in Figure 5-29. The 24-sided cylinder reflects an earlier design concept, but the thermal results are not expected to change appreciably for the 9-sided baseline design.

Adjacent nodes are coupled to each other in the vertical direction and to the neighboring heat pipe in the circumferential direction. Thermal coupling between pipes is provided at the solar cell end and it is assumed that there is no coupling between pipes or to the radiator at the cold radiator end.

Solar, albedo and Earth fluxes were computed for each node at 15 degree intervals for the elliptical and circular orbits at various beta angles. Referring to Figure 5-29, nodes 1, 6, 12, and 18 represent points on the lateral surfaces at 90 degrees increments around the circumference. Surface number 12 faces the Earth, surface number one is zenith pointing, and surface number 6 is in the direction of the velocity vector. Heat pipes are spaced on 23 cm centers and the face skin thickness is 1 mm.

Figure 5-30 is a plot of solar cell surface temperatures at 400 km circular and Beta = 0 with the sun normal to the lateral surfaces of the observatory. The observatory is Earth orientated and rotates at 1 revolution per orbit (rpo). This plot represents a hot case.

Figure 5-31 is a plot of surface temperatures at 400 km circular and Beta = 30. The orbit plane has been rotated so that the solar cell end of the observatory is tilted at a 30 degree angle to the sun. This case is significant because the peak temperature, 35 degrees C, is the maximum predicted for any of the cases studied. This solar cell temperature is of short duration and does not seriously compromise the power generation capability.

Figure 5-32 is a plot of surface temperatures at 400 km circular and Beta = 0. Cold case environmental fluxes were used.

Figure 5-33 is a plot of surface tempertures at 400 km circular and Beta = 90 degrees. This plot represents the maximum cold case for the 400 km orbit.

Figure 5-34 is a plot of surface temperatures for the highly eccentric orbit (Apogee=6000 km, Perigee=130 km) and Beta = 0. This plot represents a hot case. The absence of Earth and Albedo fluxes renders this case less severe than the corresponding 400 km orbit.

Figure 5-35 is a plot of surface temperatures for the highly eccentric orbit and Beta=90. This plot represents the cold case. The behavior of the temperatures is caused by the inputs from the Earth and albedo fluxes which occur during the near-Earth perigee passes.

Aerodynamic heating occurs during the low altitude perigees of the highly elliptic orbits of the TIMED-H spacecraft. An analysis was performed for a worst case 6000 by 130 km eccentric orbit, assuming that the observatory is orientated so that air impingement is directly on the lateral surface.

The results of the analysis are shown in Figure 5-36. For comparative purposes, the solar constant is 1352 w/m^2, about 60 percent of the maximum aerodynamic heating rate. The total energy input is not large by comparison to the solar input due to the short time duration.

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TIMED CIRCULAR ORBIT THERMAL PERFORMANCE, BETA ANGLE = 0 DEGREES, HOT CASE FLUXES

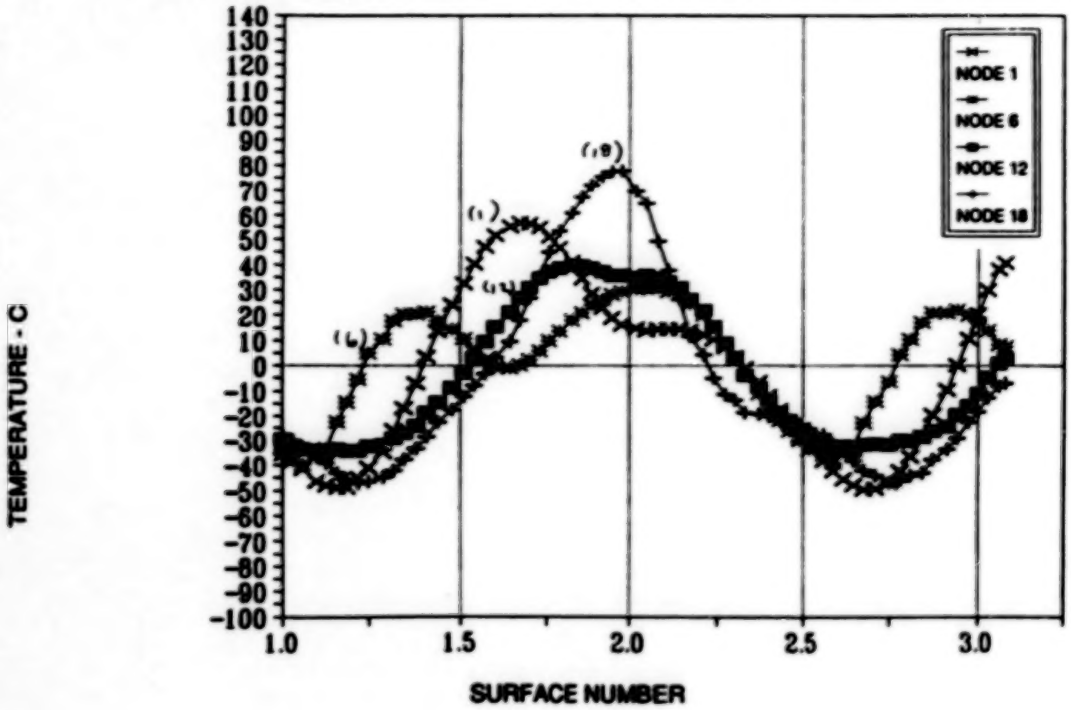

Figure 5-30

TIMED CIRCULAR ORBIT PERFORMANCE, BETA ANGLE = 30 DEGREES, HOT CASE FLUXES

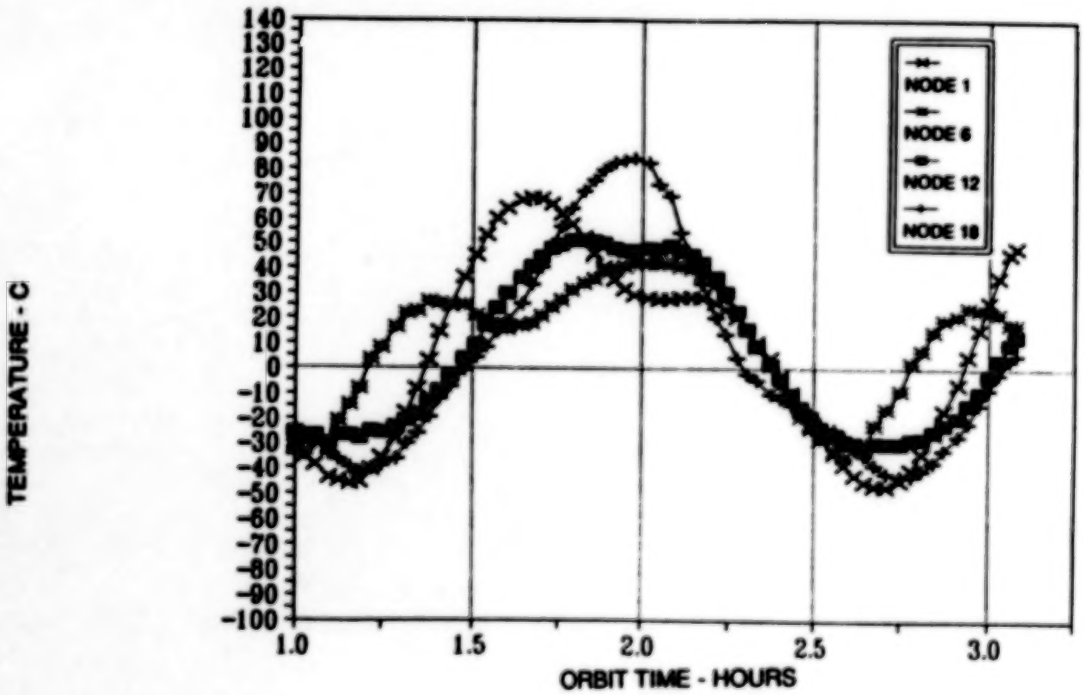

Figure 5-31

088

TIMED CIRCULAR ORBIT THERMAL PERFORMANCE, BETA ANGLE = 0 DEGREES, COLD CASE FLUXES

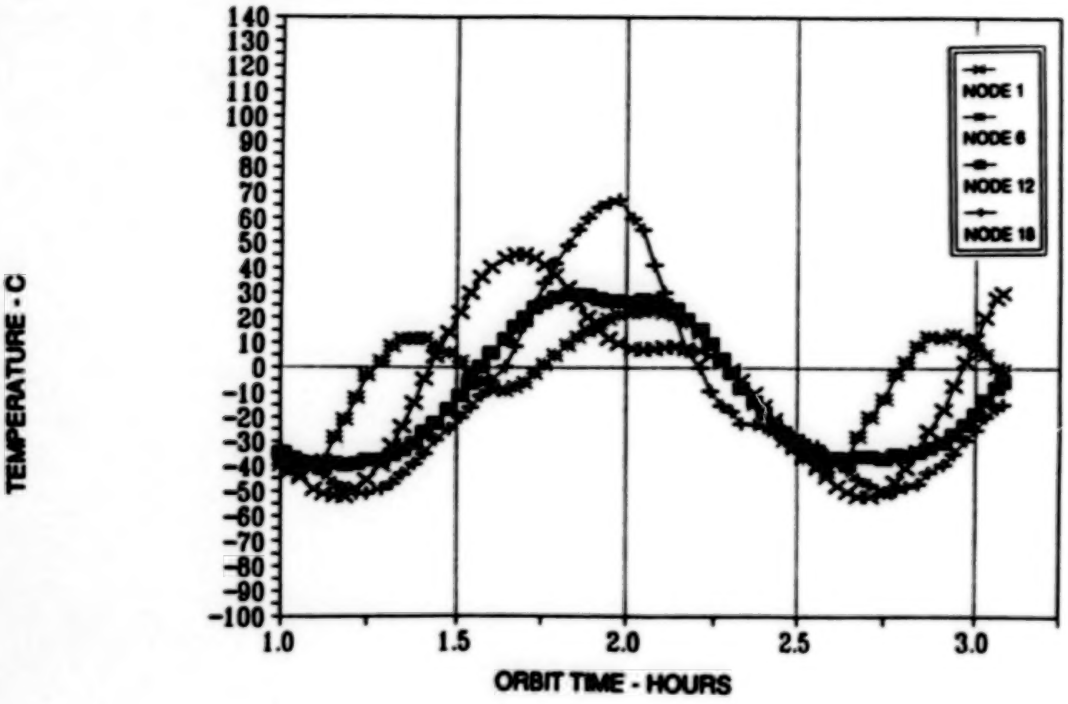

Figure 5-32

TIMED CIRCULAR ORBIT THERMAL PERFORMANCE, BETA ANGLE = 90 DEGREES, COLD CASE FLUXES

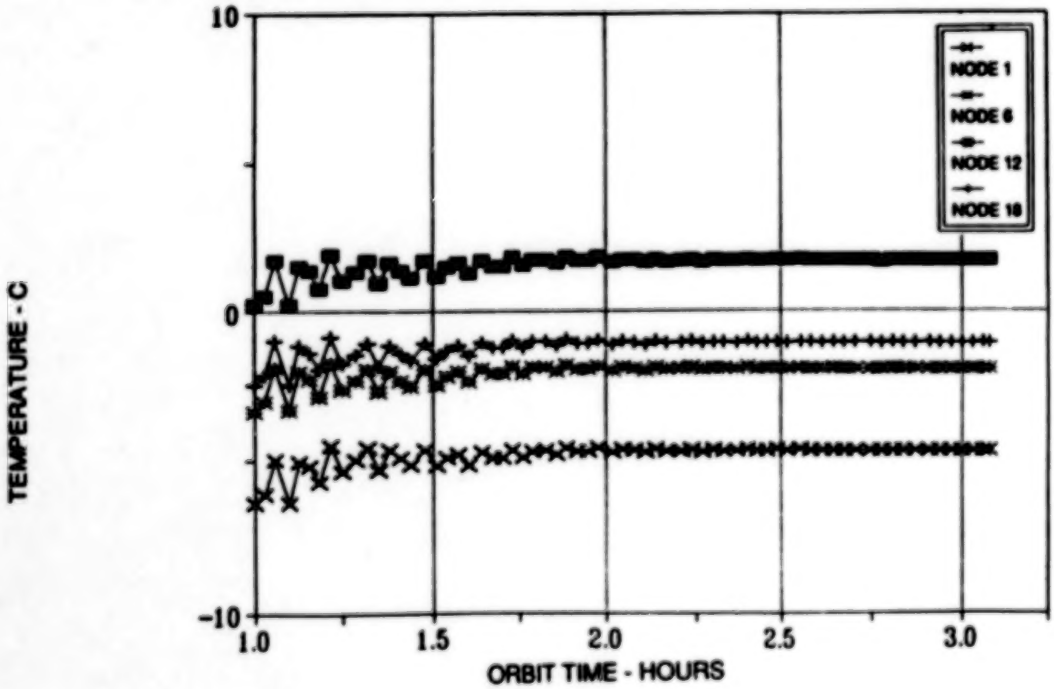

Figure 5-33

089

TIMED HIGHLY ELLIPTICAL ORBIT THERMAL PERFORMANCE, BETA ANGLE = 0 DEGREES, HOT CASE FLUXES

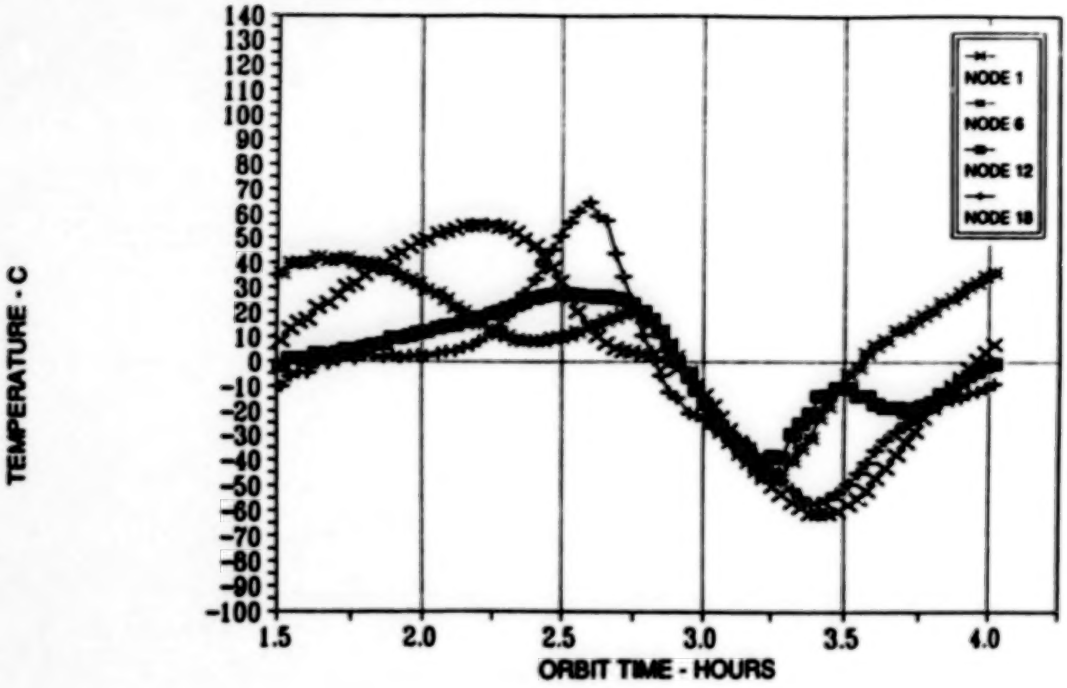

Figure 5-34

TIMED HIGHLY ELLIPTICAL ORBIT THERMAL PERFORMANCE, BETA ANGLE = 90 DEGREES, COLD CASE FLUXES

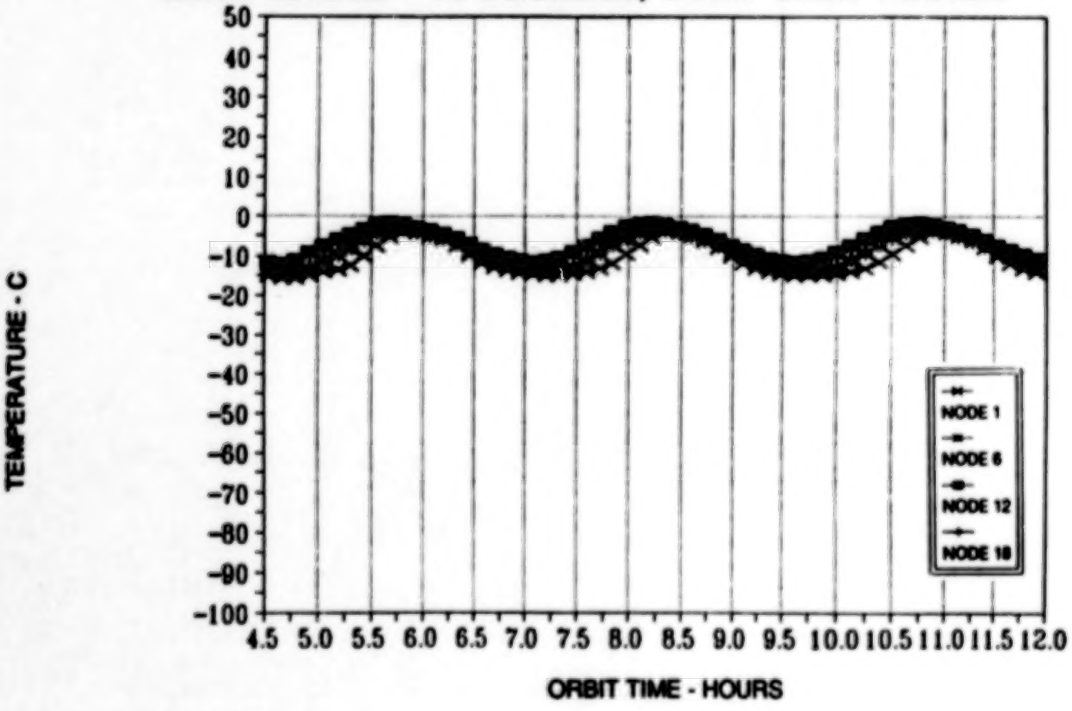

Figure 5-35

090

TIMED AERODYNAMIC HEAT RATES

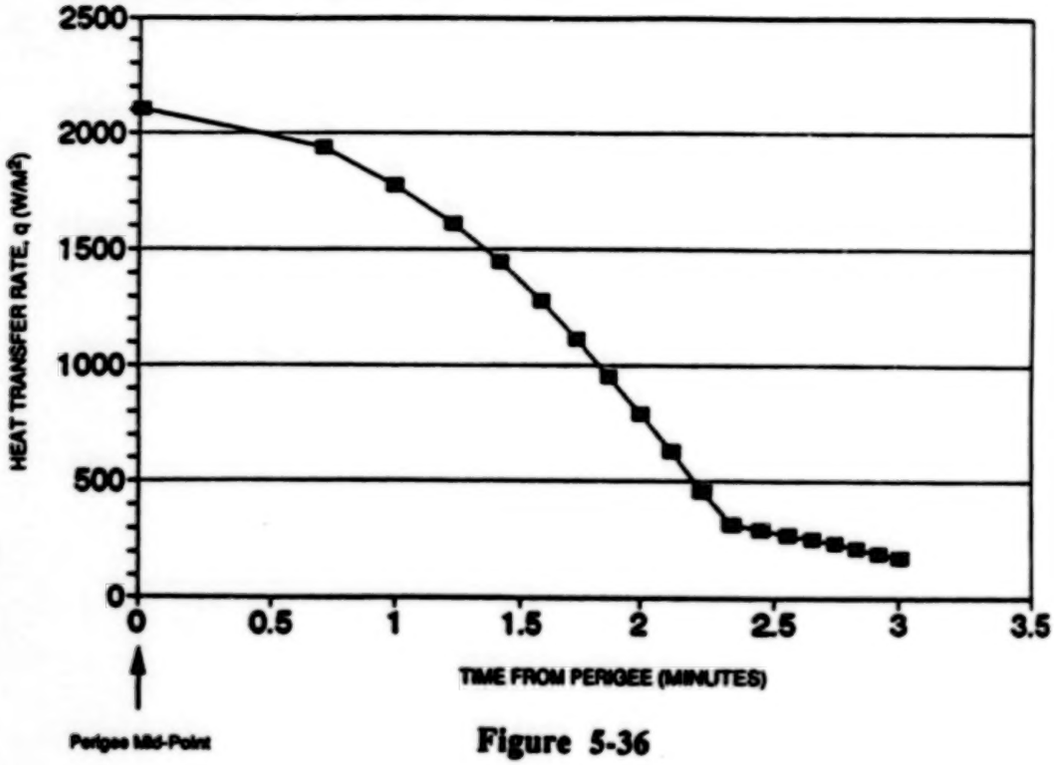

Figure 5-36

TIMED COMBINED AERODYNAMIC AND SOLAR HEATING THERMAL PERFORMANCE

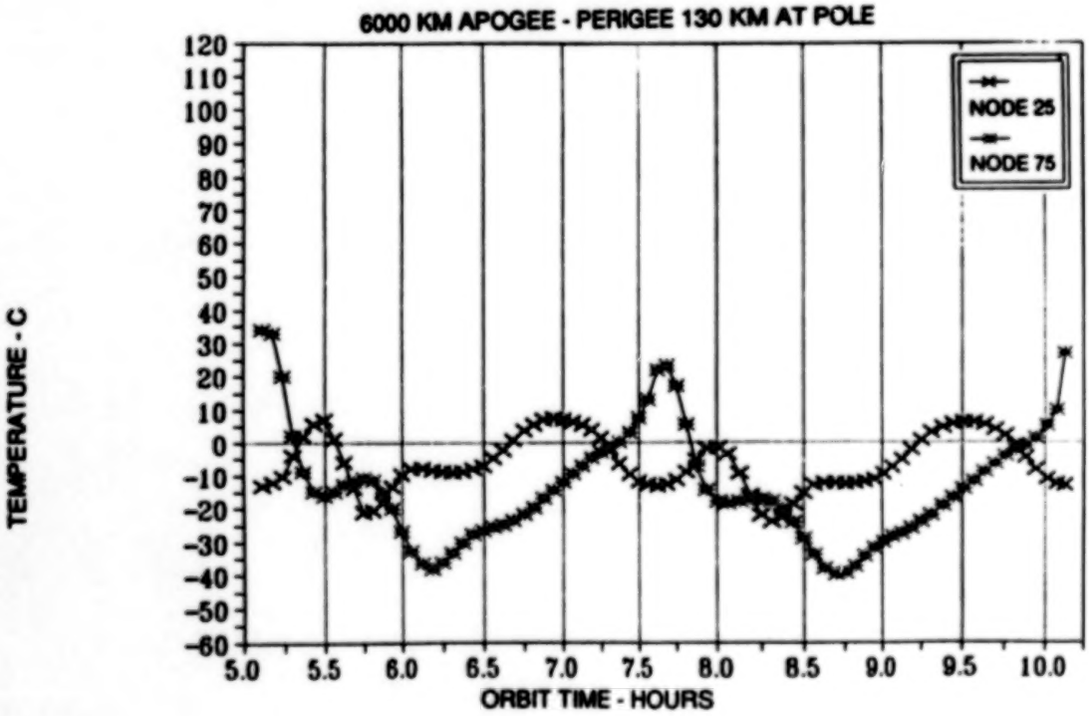

Figure 5-37

091

Figure 5-37 shows the results of a multinode SINDA analysis in which molecular heating was added to a X-axis facing surface. The temperatures are of the surface nodes between two adjacent heat pipes on 23 cm centers. The face skin thickness is 0.1 cm. Nodes 25 and 75 represent two diametrically opposite points on the lateral surface.

Even with worst case heating, it is possible to limit observatory body molecular heating to relatively small temperature excursions.

Since molecular heating occurs over a relatively short time period the temperature rise of any component is mass dependent. For example, in the case of thin walled extendable booms, it may be necessary to increase the gauge if the element is orientated normal to the flight direction.

Preliminary analysis of the deployable solar array on TIMED-L shows that a maximum temperature of 70 degrees C is reached when the array is normal to the sun during the beta = 0 orbit. The back side of this array is painted white to minimize solar input.

Internal Thermal Analysis

This section includes thermal analysis for the instruments and subsystems located inside the observatory.

A multi-node transient model was generated to evaluate the heating effects from the instrument and subsystem on their respective mounting plates. Each mounting plate is divided into 90 nodes and four circumferential heat pipes. Thermal couplings between the component baseplates and their associated footprint nodes is determined by calculating the rad-couplings and converting them to a contact conductance equivalent to 79 w/cm^2. Although, this analysis is not complete, work has progressed to the point where the following preliminary conclusions can be drawn:

Preliminary analysis indicates that instrument and subsystem temperature limitis are satisfied. Heat pipes are required in each mounting plate to limit the temperature gradients to acceptable values. Temperature differences without heat pipes can exceed 60 degrees C. With some exceptions, which may require aluminum doublers, four circumferential heat pipes can limit the inter-mounting plate temperature gradients to 9 degrees C.

A dedicated radiator for the IRLS instrument on TIMED-L maintains a -5 degrees C heat sink for the Sterling cycle cooler. The IRLS radiator is located in the center of the radiator platform, inside the area bounded by the adapter ring. An anti-sun facing area equal to 0.40 square meters rejects 50 watts at -5 degrees C during the Beta = 0 hot case when there is maximum Earth input. In the absence of active temperature control such as louvers or heaters, the heat sink temperature decreases to -17 degrees C during the cold extreme when the beta angle increases to 90 degrees. Should this lower limit prove to be unacceptable, make-up heaters will be required.

The NIRS on TIMED-H also has a dedicated radiator panel designed to maintain an upper temperature limit of -5 degrees C while rejecting 7.5 watts to space. An area equal to 0.06 m^2 is required for the NIRS.

A thermal trade study was conducted to evaluate the feasibility of eliminating the yaw turn maneuver. The benefit of eliminating this maneuver is preserving the preferred instrument orientation for all instruments and thereby increasing the total science observation time. In

order to reject heat from either end of the observatory, two thermal control heat transport paths must be provided. Each must act as a thermal switch to isolate components from the "hot end" radiator when it is in sunlight. Variable conductance heat pipes and/or capillary pump loops can be used to provide the necessary isolation while providing adequate heat transport during the normal mode. However, these devices would require early development and performance verification testing. This alternate design implementation would result in a significantly more complex and heavier subsystem.

An analysis was performed to show that the thermal subsystem hardware items are capable of operation when the spin rate of the spacecraft is 4 rpm. Since the force at the maximum moment arm is less than 0.1 g, neither the louvers or heat pipes are adversely affected.

5.6.4 Reliability, Heritage and Risk Assessment

The survivability factor for the TCS is estimated at 0.997 after two years and is reasonable for the semi-active thermal control subsystem (TCS) using all flight-proven hardware and possessing significant heritage from past satellites.

Major component heritage is summarized in Table 5-8.

TIMED THERMAL SUBSYSTEM COMPONENT HERITAGE

COMPONENT	HERITAGE
Heat Pipes	ATS-6, HST, TOPEX, SPARTAN 201
Louvers	Mariner, ATS, NIMBUS, OAO

Table 5-8

Location of heaters and thermostats can minimize the effect of a single component failure. Thermal louvers provide multi-fault tolerance since each blade is independently actuated by a bimetallic spring and the failure of any individual blade has a small effect on overall subsystem performance. Heat pipes share the thermal load and have inherent redundancy. A major advantage of the heat pipe matrix is the increase in the effective system thermal time constant.

Preliminary failure mode effects analysis is described in Table 5-9.

TIMED THERMAL SUBSYSTEM COMPONENT FAILURE EFFECTS ANALYSIS

FAILURE	BACKUP MODE	MISSION IMPACT
Heaters	Redundant heaters	Small temperature difference expected
Thermostats	Series and parallel circuit redundancy	None
Heat pipes	Multiple heat pipes provide redundancy	Small degradation of thermal performance
Louvers	Each blade independently actuated	Small performance degradation if one or two blades become inoperative.

Table 5-9

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5.7 Command and Data Handling Subsystem

5.7.1 Requirements

The command and data handling (C&DH) subsystem satisfies the following requirements:

- Decode the Consultative Committee for Space Data Systems (CCSDS) uplink commands received from the Wallops Orbital Tracking Station (WOTS)at 2 kbps and from Tracking and Data Relay Satellite (TDRSS) at 1 kbps.
- Distribute realtime CCSDS command messages and/or store command messages in memory for later delivery.
- Acquire and store science, housekeeping, performance, ancillary and status data in mass memory.
- Provide sufficient mass memory storage for 24 hours of housekeeping and instrument telemetry data based on 100 percent instrument duty cycle plus 25 percent margin.
- Provide simultaneous telemetry data record and playback capability.
- Provide processors to host flight software.
- Provide the capability to load flight processors.
- Format downlink data using CCSDS transfer frame protocols. Provide realtime health and safety telemetry data transfer frames at 1 kbps and playback data at 1.5 or 3.0 Mbps.
- Maintain the spacecraft clock and distribute time references.

5.7.2 Subsystem Description

The C&DH subsystem block diagram is shown in Figure 5-38. The major components are the communications processor, the spacecraft processor and dual-redundant MIL-STD-1773 fiber-optic data buses.

The following paragraphs describe the functions and capabilities of each C&DH subsystem component.

The TIMED C&DH architecture is a redundant, distributed data bus system that allows instruments and spacecraft subsystems to be connected as a local area network. A trade study comparing a standard bus and point-to-point C&DH architectures found that the use of a standard bus architecture reduces the size, weight, power and complexity of the spacecraft harness, simplifies the interfaces, reduces the instrument integration and test effort, and simplifies the incorporation of instrument configuration changes.

Communications Processor

The communications processor consists of redundant processors, redundant low-voltage power converters, redundant downlink cards, and solid state mass memory. Each processor card contains an 80C86 processor, an 80C87 coprocessor, an 82C37A Direct Memory Access (DMA) bus controller, read-only memory (ROM) for booting up the processor, programmable ROM (PROM), electronically erasable PROM (EEPROM),

random access memory (ram), an Application Specific Integrated Circuit (ASIC) that controls the backplane bus, and two MIL-STD-1773 bus controller/monitor terminal units.

COMMAND AND DATA HANDLING SUBSYSTEM BLOCK DIAGRAM

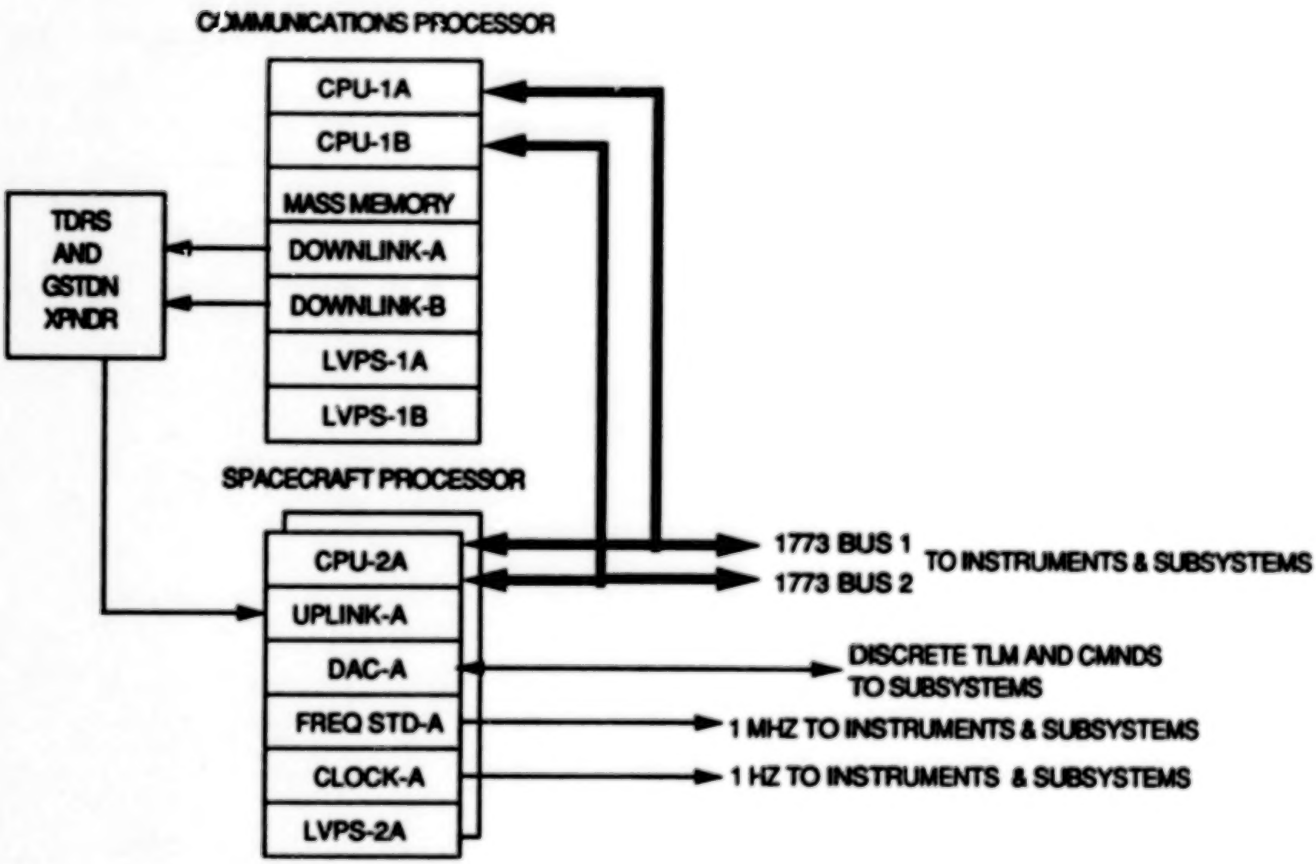

Figure 5-38

The communications processor coordinates all high rate or high volume communication functions on board the observatory. One of its MIL-STD-1773 terminal units is the instrument data bus controller and the other is the spacecraft data bus remote terminal and monitor. Command mode blocks received from the ground are processed by the spacecraft processor and, if appropriate, are sent to the communications processor via the spacecraft data bus. The communications processor distributes appropriate real-time commands to the spacecraft subsystems and instruments via the instrument and spacecraft data buses and stores command loads scheduled for later execution. The communications processor is also responsible for acquisition of health and safety, housekeeping, and science telemetry data from the instruments, and the spacecraft subsystems via the data buses. The communications processor packetizes data that it receives in unpacketized form from the subsystem, and optionally from the instruments. The processor then passes the data to the bus control ASIC via the processor local bus. The ASIC formats the data packets into CCSDS Virtual Channel Data Units, adds error dectection and correction (EDAC) parity

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bits to each 32 data bits, and writes the data to the mass memory via the backplane bus. Periodically (seconds or minutes), the ASIC reads the stored data, and if in error, computes new parity bits, and rewrites the data back into the mass memory. When data is retrieved from the memory for downlinking to the ground, the EDAC parity bits are stripped off so they do not affect the downlink transmission rate or downlink contact time requirement.

The solid state mass memory consists of four 670-Mb memory cards to meet the requirement of 2.6 G-bits of usable memory requirements. Each of these cards, containing ram chips and a memory controller, is connected to the communications processor and downlink cards via the backplane. Solid state memory is lighter, cheaper, more reliable, more power efficient, and more flexible than magnetic tape recorders. Solid state memory has no moving parts, provide a simultaneous record and playback capability, and any portion of memory can be accessed directly for retransmission in case of partial loss during playback.

Redundant downlink cards are cross-strapped to the STDN and TDRSS transponders to provide the interface between the C&DH and the communication subsystems. The downlink card provides a communications path between the communications processor and the transponders for status data acquisition and distribution of commands and clock signals. During contacts with TDRS, the downlink card receives real-time health and safety telemetry data transfer frames from the communications processor at a 1 kbps rate. The downlink card optionally, convolutionally encodes the transfer frames and passes the data to the TDRSS transponder for transmission to the ground. During contacts with WOTS, the downlink card receives playback data at a selectable data rate of either 1.5 Mbps or 3 Mbps. In turn, the downlink card passes the transfer frames to the STDN transponder.

The low voltage power supply (LVPC) cards provide power conversion from the spacecraft's -28 V unregulated power bus to the low-voltage levels required by the communications processor. Only the primary side of the communications processor section and the mass memory are powered on. The redundant communications processor, LVPC and downlink cards are powered off. Software and hardware watchdog capabilities are provided to monitor the status of the communications section and to initiate resets or switch overs to the redundant cards.

Spacecraft Processor

The redundant spacecraft processor consists of a spacecraft processor, a data acquisition card, an uplink card, a clock and frequency standard card, and an LVPC. The spacecraft processor is similar to the communications processor except for higher end processors and additional memory and contains an 80C386 processor, an 80C387 coprocessor, and an 82380 DMA bus controller, ROM for booting up the processor, PROM, EEPROM, ram, an ASIC that controls the backplane bus, and two MIL-STD-1773 bus controller/monitor terminal units.

The spacecraft processor provides the primary attitude control subsystem (ACS) processing capability, communicating with the ACS electronics and actuators via messages over the spacecraft bus. The spacecraft processor also provides the logic and control functions for the data acquisition card. One of the two bus terminal units is the spacecraft data bus controller, and the other is the instrument data bus remote terminal and monitor.

The data acquisition card, in conjunction with the spacecraft processor, provides a discrete telemetry and command interface. The spacecraft processor provides the logic and control capabilities for the data acquisition card. It is also responsible for packetizing the telemetry data and sending the packets to the communications processor for storage. The telemetry

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portion of the data acquisition card acquires analog, serial digital and bi-level telemetry. The command portion of the data acquisition card distributes discrete and serial digital commands, serial digital enables, clock signals, and -28 V pulse lines.

The uplink card handles the command uplink signals received by the transponders, decodes the message and passes it to the spacecraft processor.

The clock card generates clock signals required by the C&DH processors, spacecraft subsystems and instruments from the frequency standard. The clock card broadcasts a logic "ON" synchronization pulse once per second via a time synchronization wire-line bus and also maintains Coordinated Universal Time. The spacecraft processor broadcasts the time message to both the data buses. These messages indicate the time at the occurrence of the next 1 Hz synchronization pulse. Time is encoded in accordance with CCSDS Recommendation 301.0-B-2, Time Code Formats, using the CCSDS Day Segmented Time Code millisecond option. There are provisions to correct the spacecraft clock through ground commands.

The LVPC cards are identical to those specified for the communications processor.

MIL-STD-1773 Bus

The MIL-STD-1773 fiber-optic bus consists of separate transmit and receive optical cables from each of up to 16 bus users connected through a star coupler. The star coupler fuses together all 16 input and output cables, such that a message from any bus user is received by all users. User access to the bus is via a terminal unit that consists of a redundant fiber-optic transceiver, a MIL-STD-1553B protocol unit, and miscellaneous logic and signal conditioning hardware. Terminal units are either remote terminals or bus controller/monitor/remote terminals used in the processor.

The C&DH has two independent 1773 buses, each having a redundant, physical, fiber-optic network. Users configured as a node on either, or both, buses have a redundant, physical, fiber connection with the bus, consistent with the 1773 protocol. Should a failure occur on any single fiber or single transmitter/receiver, the redundant fiber/transmitter/receiver would automatically take over. The configuration for this type of connection is shown in Figure 5-39 below.

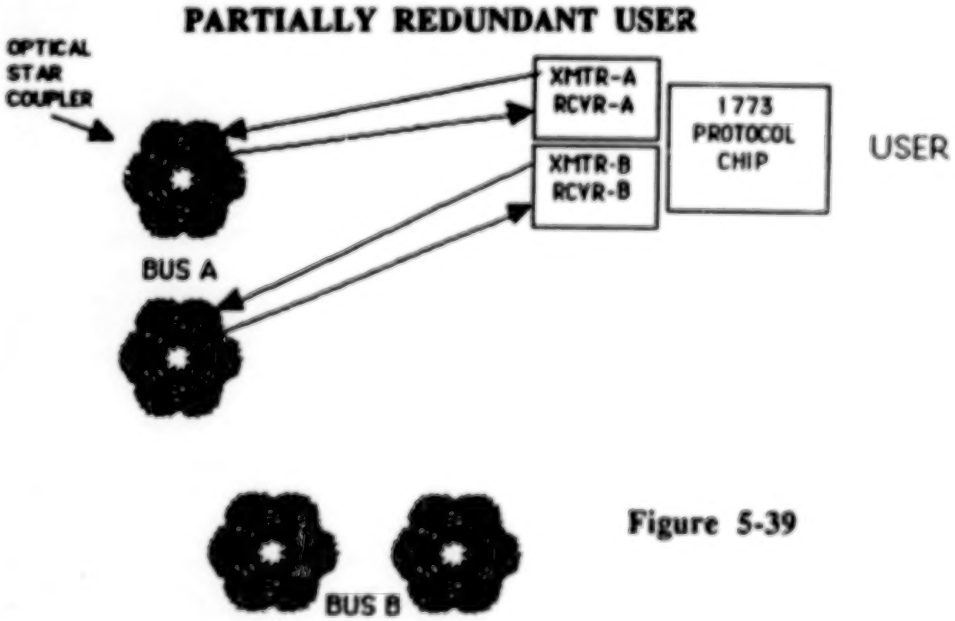

Figure 5-39

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A user who would need full redundancy can be configurated with a node on both 1773 buses, although not simultaneously. The user would be assigned a separate address on each bus and would require another complete set of interfacing hardware along with another protocol unit. In this configuration, should a failure occur on any protocol unit, the optional redundant protocol unit would be activated on the alternate bus. The configuration for this type of connection is shown in Figure 5-40 below.

FULLY REDUNDANT USER

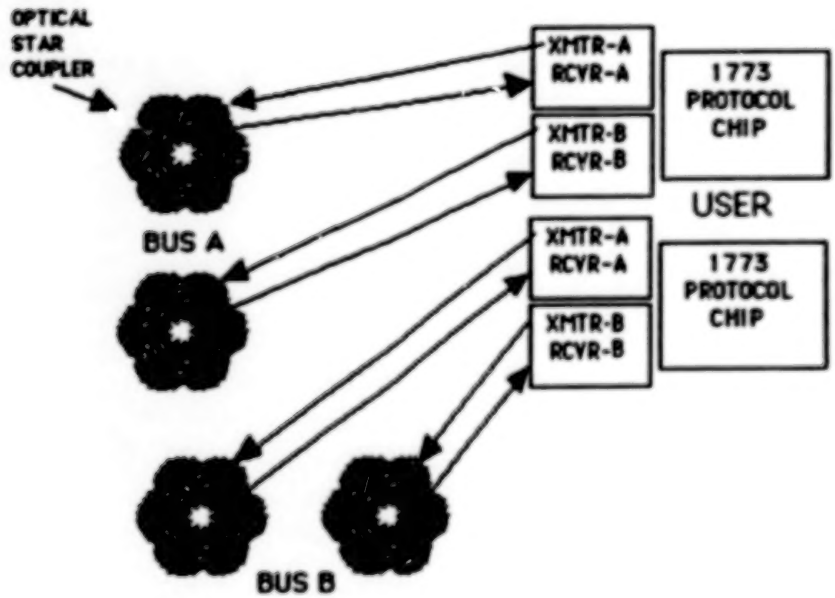

Figure 5-40

The function of the bus controller is to control all communications over the bus so that only one terminal transmits at any given time and to monitor the status of all the remote terminals. The function of the bus monitor is to take over the bus controller function in case of failure of the bus controller until the backup processor is booted up. The input/output characteristics of the data buses are listed in Table 5-10.

TIMED SPACECRAFT DATA BUS CHARACTERISTICS

Transmission Method	Half-Duplex, Asynchronous
Bit Format	Manchester II Bi-Phase Encoding
Bit Rate	1 Mbps
Word Length	20 bits (3 synch, 16 data, 1 parity)
Word Formats	Command words, Data words, and Status words
Transfer Protocol	BC to RT, RT to BC, RT to RT

Table 5-10

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5.7.3 Performance and Analysis

The solid state mass memory was sized to accommodate 24 hours of housekeeping and science telemetry data based on 100 percent instrument duty cycle. Ten percent additional memory was added for packet and transfer frame headers and an additional 25 percent margin was added. The resulting memory sizes are 2.4 Gb for the TIMED-H and 1.4 Gb for the TIMED-L.

5.7.4 Reliability, Heritage and Risk Assessment

The probability of success of the C&DH subsystem based on reliability values for the SAMPEX 80C386 and 80C86 processors, uplink/downlink cards, mass memory backplane and LVPC's is 0.953. Preliminary failure mode and effects analysis is shown in Table 5-11.

The TIMED C&DH subsystem is based on the SEDS implementation being flown on the Solar Anomalous and Magnetospheric Particle Explorer, and variations proposed for the X-Ray Timing Explorer (XTE) and the Tropical Rainfall Measuring Mission (TRMM). Table 5-12 summarizes the heritage of the C&DH components.

TIMED C&DH SUBSYSTEM FAILURE ANALYSIS

Failure	Backup Mode	Mission Impact
Uplink Card	Use redundant card	Loss of command link until failure is detected and backup is turned on.
Clock Card	Use backup clock card and frequency standard	Instruments and subsystems clocks will free run and drift until backup clock takes over. Drift should not be significant
Frequency Standard	Use backup clock card and frequency standard	Same as above.
Spacecraft Processor	Use redundant processor card	Communications processor takes over as spacecraft bus controller with little or no impact. Attitude control subsystem computations cease until backup unit boots up. Flight software must address recovery.
Communications processor	Switch to redundant processor	Telemetry data acquisition would be disrupted until backup processor is booted up. Spacecraft processor would take over as instrument data bus controller temporarily. Uplink and downlink operations would be disrupted if failure occurs during a contact.
Downlink Card	Switch to redundant card	Downlink contact would be disrupted until redundant card is activated.
LVPC Card	Switch to redundant LVPC card, activating the backup cards connected to the failed LVPC card	C&DH section powered by the failed LVPC card will be disrupted until backup cards are powered up.
Memory Card	Detect memory loss and stop using that area	Data stored in failed area would be lost. The storage capability of that partition of memory would have reduced capability.
Data Bus	Bus controller commands all bus terminals to use the redundant bus	Bus communications would be disrupted for up to a major frame while the switch over takes place. Impact should be minimal.
Remote Terminal	Bus controller switches all bus terminals to the backup bus	Affected instrument or subsystem would lose bus communications until the switch over to the redundant bus.

Table 5-11

TIMED C&DH SUBSYSTEM COMPONENT HERITAGE

Component	Heritage
Uplink, Clock, Frequency Std. and Memory Cards	TRMM
Spacecraft and Communications Processors	SAMPEX, TRMM
Downlink Card	SAMPEX, TRMM
Fiber-Optic Data Buses	SAMPEX
Remote Terminals	SAMPEX, TRMM

Table 5-12

5.8 Flight Software

5.8.1 Flight Software Requirements

The flight software shall provide the following:

- An operating system which performs initialization, applications services, memory management and error handling,

- Applications services to instruments and subsystems.

5.8.2 Flight Software Description

The flight software is described in the functional areas of operating system, spacecraft system and subsystems, telemetry and command processing, and instrument services.

Operating System

The operating system software provides for multi-tasking; event driven, priority-based task scheduling; intertask communication and synchronization; real-time clock support; memory management; and data error detection. Virtual Realtime Executive (VRTX) commercial off the shelf (COTS) software is baselined.

Spacecraft System and Subsystems

The spacecraft system and subsystems software provides coordination among subsystems for spacecraft spin-up for instrument calibration, for low perigee operation, and for yaw maneuvers; manages thruster scheduling and instrument safing for Delta-V burns; collects subsystem and instrument data for trend analysis of observatory operations; performs fault management and system level diagnostics, including transmission of diagnostic data to the ground; and issues spacecraft safing commands when anomalous conditions are detected.

Spacecraft monitoring software performs limit checking of subsystem telemetry and monitors the performance of the power subsystem and the thermal subsystem. The software monitors the state of the command and data handling subsystem hardware and software and places the spacecraft in safehold or takes other appropriate action upon detection of errors.

The attitude control software establishes acquisition of initial attitude control upon detection of separation from the launch vehicle, applies filters to the sensor data for use by the control algorithms, maintains the observatory orbit position, commands the ACS actuators according to the control algorithms, models sun position, implements observatory dipping control algorithms, executes the thermal yaw maneuver, and implements safe ocean disposal at end of the mission.

The propulsion control software provides control of the orbit adjust thrusters and manages tank propellent to maintain coarse center of gravity.

Telemetry and Command Processing

The telemetry processing software collects science and housekeeping data from the spacecraft subsystems and instruments on either a scheduled or an asynchronous basis; formats the spacecraft telemetry into CCSDS packets, including ancillary data, time and

header information; stores science and housekeeping data on the C&DH solid state recorder; and outputs the telemetry to the communications subsystem. This software implements #1773 bus protocol.

The command processing software accepts and processes spacecraft and instrument CCSDS command packets and checks the commands for transmission errors. This software validates all commands before execution and requests ground confirmation for hazardous or unreasonable (e.g., fire opposing thrusters) commands and distributes valid command packets to the appropriate application module or instrument. In addition, this software provides scheduling of time tagged stored command sequences and verifies memory loads by checksum or other means prior to overwriting memory contents. The status of commands received and verified are logged and downlinked in telemetry.

Instrument Services

This software activates and deactivates instruments in accordance with schedules, coordinates science observations among multiple instruments if needed, issues instrument safing commands when anomalous conditions are detected, and provides computational services to the instruments if required.

The flight software size estimate of 15,600 source lines of code (SLOC) was based on studying system and subsystem software requirements, as well as on comparison with the flight software implementation of the Small Mesospheric Explorer (SAMPEX), X-Ray Timing Eplorer (XTE) and the Tropical Rainfall Measuring Mission (TRMM) (which use the same C&DH as TIMED) and the Extreme Ultraviolet Explorer (EUVE) and the Upper Atmosphere Research Satellite (UARS) (which perform many of the same functions). The software sizing estimate breakdown shown in Table 5-13.

FLIGHT SOFTWARE SIZING ESTIMATE

OPERATING SYSTEM SOFTWARE	COTS
APPLICATIONS SOFTWARE	
INSTRUMENTS	200
SYSTEMS	600
COMMUNICATIONS	150
COMMAND & DATA HANDLING	8,600
ATTITUDE CONTROL SYSTEM	4,450
PROPULSION	950
POWER	500
THERMAL	150
TOTAL SOURCE LINES OF CODE	15,600

Table 5-13

Flight Software Development Facility

The flight software development facility contains hardware and software, including COTS software tools, which provides an environment for the development, test, integration and maintenance of the flight software. The software is developed on a computer such as a Micro VAX or a Sun Workstation and is tested using computers which are functionally

equivalent (FEU), single-string, ground versions of the flight-qualified C&DH computers. The facility also contains editors, compilers and linkers needed to generate load images.

Test software is required for development and test of the flight software. The spacecraft integration and test (I&T) electrical ground support equipment (EGSE) software will provide Mission Operations Control Center (MOCC) capabilities, such as ground command generation and telemetry display, needed to interface to the flight software during test.

A software spacecraft simulator provides simulation of the flight software external interfaces, such as the interfaces with the attitude control system hardware, sensor outputs to the flight software and responses to commands from the flight software. The simulator software has been sized at 10,000 SLOC.

5.8.3 Performance and Analysis

The flight software has adequate growth margins for CPU, CPU memory and 1773 bus utilization. Margin calculations are based on a single software load concept; i.e., identical software images loaded into each of the four computers (two 80386, two 8086) on each spacecraft. The growth margins are shown in Table 5-14.

FLIGHT SOFTWARE MARGINS

	Spacecraft (80386) Processor			Communications (8086) Processor		
	Capacity	Utilization	Margin	Capacity	Utilization	Margin
CPU	3 MIPS	35%	65%	.3 MIPS	40%	60%
CPU Memory	1 MBYTE	50% (1)	50%	.5 MBYTE	60% (1)	40%
1773 Bus	1.0 MBPS	3%	97%	1.0 MBPS	3%	97%

(1) UTILIZATION BASED ON 20 BYTES PER SOURCE LINE OF CODE, STORED COMMANDS AND MESSAGE BUFFERS (UARS)

Table 5-14

During spacecraft integration and test, the software is hosted on the flight computers, with the EGSE providing MOCC capabilities. After launch, the flight software is maintained by a software development and verification facility (SDVF) located in the MOCC. Software loads and patches are developed and tested using the development computer, the C&DH FEU computers and the software simulator, then uplinked to the spacecraft C&DH.

5.8.4 Reliability and Heritage

Reliability is provided by routine error checking (e.g., noisy sensor data, transmission errors, limit checking of sensor input data and actuator output commands); software self-tests which trigger hardware safehold if critical software errors are detected; and software reprogramming from ground to work around hardware or software failures.

Flight software development, including design, coding, test and documentation adheres to applicable NASA, GSFC and TIMED Project standards. Software requirements are obtained from systems requirements as defined by systems engineering.

The TIMED flight software design is based on the Small Explorer Data Systems (SEDS) flight software as implemented for the SAMPEX, XTE and TRMM Projects. After the TIMED requirements have been defined, the SEDS software will be analyzed to determine how much of the design and code can be reused for the TIMED flight software, how much

new or modified code will be required to incorporate unique TIMED requirements, and how much code may be discarded as not applicable. The COTS VRTX operating system used by SEDS has been baselined for TIMED.

5.9 Communications Subsystem

5.9.1 Requirements

The communications subsystem satisfies the following requirements:

- Provide command uplink through the Tracking and Data Relay Satellite System (TDRSS) and the Wallops Orbital Tracking Station (WOTS)
- Provide science data downlink through WOTS and health and safety plus ranging through TDRSS.
- Downlink 48 hours of stored data during a consecutive 48-hour period to WOTS.

5.9.2 Subsystem Description

The communications subsystem consists of one TDRSS transponder and one GSTDN transponder, band reject filters, diplexers, a hybrid, an RF switch with load, and two omni antennae. As seen in the system block diagram, Figure 5-41, the transponders, band reject filters, diplexers, and antennas are redundant. The antenna coupling devices are passive and are not redundant. The RF switch, used for antenna selection, is not redundant and will normally be left in the "Earth favored" antenna position. Power ON/OFF to the transponders, in addition to power ON/OFF to the transmitters is controlled by the C&DH subsystem.

Omni Antennae

The S-band omni antennae, which are manufactured by Ball Aerospace, are located on the plus and minus Y-axis, tilted 15 degrees from the Z-axis, at a location as close to the ±Z-axis as possible. Figure 5-42 shows the antenna placement.

Transponders

Both the GSTDN and TDRSS transponders contain: a transmitter with a modulator and telemetry encoder; a receiver with a demodulator and command decoder; a "turnaround" channel to move the ranging signal from the received signal and modulate it on the transmitted signal; and monitor and control circuitry. The major differences of the transponders are discussed below.

GSTDN Transponder - This transponder, which is manufactured by Motorola, is designed for ground support only and will not support any of the TDRSS signals. The uplink command function is a 2 kb signal PSK modulated on a 16 KHz subcarrier, which is phase modulated on the uplink carrier (PCM/PSK/PM). The downlink telemetry is bi-phase modulated on the down link carrier. The ranging signal is phase modulated directly on the carrier.

TDRSS Transponder - This transponder, which is manufactured by Motorola, is designed for dual use, both TDRSS and ground, with automatic mode select based on the received power. The received power from a ground transmitter is approximately 50 db greater than that received from the TDRSS making the mode select very reliable. The mode select is commandable to remain in either mode. The transponder has all of the capabilities of the GSTDN, plus the capability to accept the quadra-phase modulation and pseudo-random noise codes of the TDRSS spread spectrum signal. The forward link has one channel for command and a separate channel for ranging. The return link has two channels for telemetry. The two channels may be modulated with the same data or with two independent data streams.

TIMED RF COMMUNICATIONS SUBSYSTEM BLOCK DIAGRAM

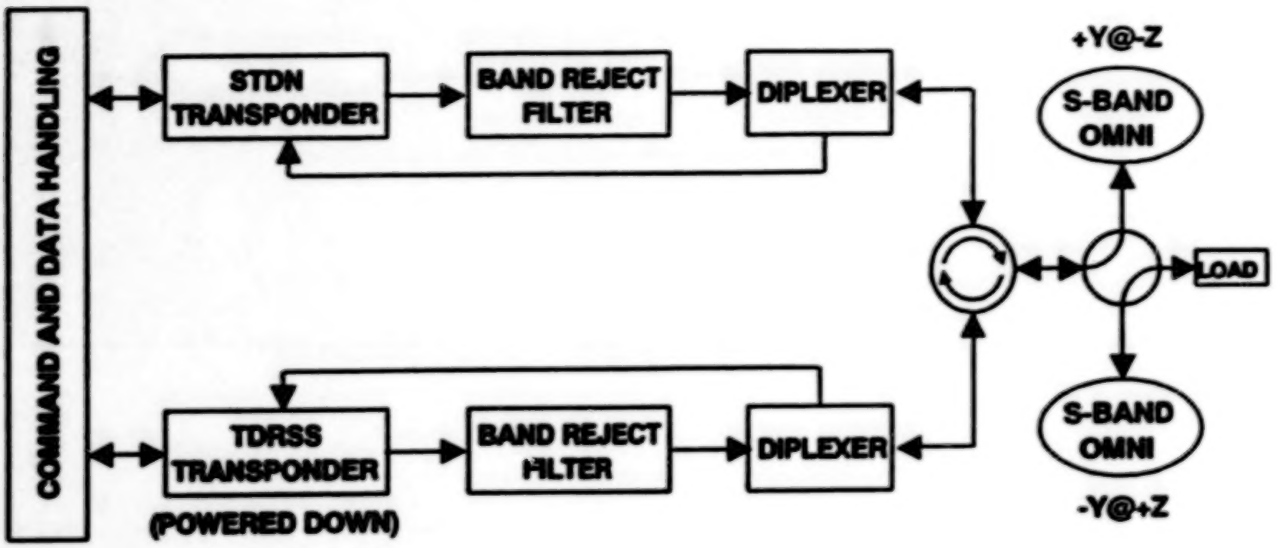

Figure 5-41

TIMED RF COMMUNICATIONS SUBSYSTEM BASELINED ANTENNA PLACEMENT

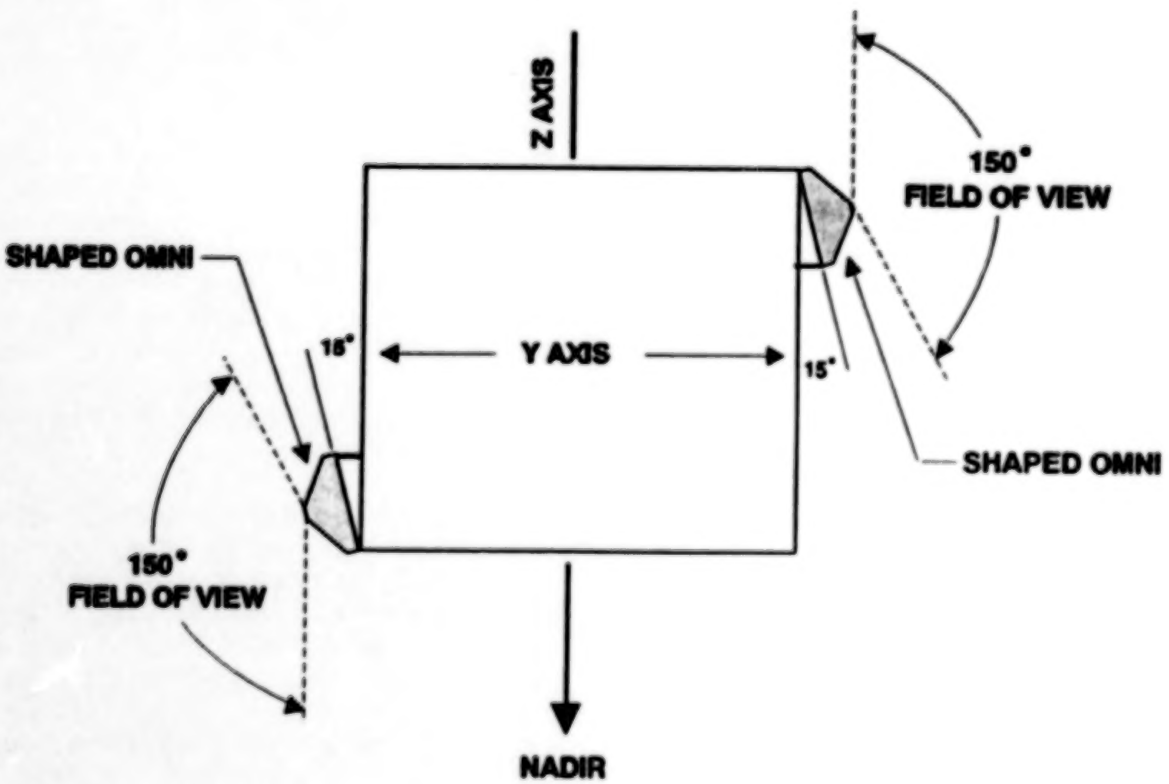

Figure 5-42

The combination of the GSTDN transponder, with the TDRSS transponder, provides minimum power, the required redundant communication path, and quick access to the observatory in a contingency situation. TDRSS support for ranging is estimated at 10 minutes per orbit. WOTS also provides ranging capability to supplement TDRSS tracking.

The transponder status is provided to the spacecraft C&DH subsystem in four formats which include analog voltage, temperature sensors, bi-level digital status and serial digital format.

Band Reject Filter

The band reject filter is a passive device that passes the frequency of the transmitter and rejects the frequency of the receiver by attenuation and is placed in the transmitter path prior to the diplexer to eliminate any spurious signals in the receive band.

Diplexer

The diplexer is a frequency selective device that is used to route the transmit and receive frequencies between the transponder and the antenna. The diplexer passes the transmit and receive frequencies to and from the antenna with no attenuation, but provides high isolation between the transmitter and receiver.

Hybrid

The hybrid is a reverse field circulator that allows multiple devices to be connected to a single port without mutual interference. It passes the transmit and receive frequencies in opposite directions.

RF Switch

This is a coaxial switch that allows antenna selection. The switch is controlled by the C&DH subsystem.

5.9.3 Performance and Analysis

RF link analysis for WOTS was performed for the worst case condition which occurs on the TIMED-H Mission during both the elliptical and circular phases. During the elliptical phase, the slant range reaches a maximum of 10,286 km and during the circular phase, the daily view time drops to less than 15 minutes requiring a 3.0-Mbps data rate.

The downlink analyses, shown in Tables 5-15 amd 5-16, indicate a 4.3 db positive margin at a 10,286 km slant range with a 1.5 megabit data rate, and a 16.1 db positive margin for the circular orbit with a 1851 km slant range and a 3 Mb data rate. An uplink RF link analysis, shown in Table 5-17, was performed for a 10,286 km slant range and found to be 58.2 db positive. A link analysis for TIMED-L was not performed since the TIMED-H circular phase had positive margins and TIMED-L has longer view periods.

The TDRSS S-band single access link analysis, shown in Tables 5-18 and 5-19 was performed for a 1.0 kbps forward and return link and found to be 6.2 db positive for the forward link and 10.6 db positive for the return link.

The TIMED-H requires 26 minutes transmit time at a 1.5 Mbps data rate and 13 minutes at a 3.0 Mbps data rate to downlink 24 hours of stored data based on 100 percent instrument power duty cycle. The TIMED-L mission will require 15 minutes transmit time at a 1.5 Mbps data rate and 8 minutes at a 3.0 Mbps data rate.

An orbital analysis to determine daily contact times was performed. A 5 degree elevation station mask and a one minute to establish contact was assumed. The orbit for TIMED-H was propagated in 24-hour periods from 85 degrees North perigee to 85 degrees South perigee in approximately 15 degree steps. For the circular phase, the orbit was propagated in 10 day periods from 128 degrees right ascending node through 360 degrees and back to 128 degrees. A review of the contact times shows that the requirement of downlinking 4.8 Gb in a 48-hour period can be met with a data rate of 1.5 Mbps, except for a short period transitioning from elliptical to circular orbit. The instrument power duty cycle during this transition period are power limited to 50 percent or less and all of this stored data can be transmitted using a 3.0 Mbps rate. The data are shown in Table 5-20 and include all acquisition times.

The orbit for TIMED-L was propagated in 2-week periods from 311 degrees right ascending node through 360 degrees back to 311 degrees five times to obtain a representative set of contact periods. The data, including acquisition times, are shown in Table 5-21.

As shown in Table 5-21, a data rate of 1.5 megabits will satisfy all mission requirements except for the TIMED-H Mission, 400 km circular Phase II. This phase requires a data rate of 3 Mb to meet the requirement of downlinking 4.52 Gb in 48 hours.

5.9.4 Reliability, Heritage and Risk Assessment

The survivability factor for the communications subsystem is estimated at least 0.9998. Of the 16 TDRSS transponders flown for 5 years or longer, only one transmitter failure occurred after 10 years of operation. The GSTDN transponders have been in use since 1965 and have experienced no failures.

A preliminary failure modes effects analysis for the communication subsystem is shown in Table 5-22.

The TDRSS transponder first flew on the SMM Mission launched in 1980. A total of 16 first-generation transponders were flown through 1984. Of these transponders, flown for 5 years or longer, only one transmitter failure occurred after 10 years of operation. A total of four, second-generation transponders have been flown on COBE and GRO. GRO has had a problem with receiver automatic gain control stability.

The GSTDN transponders have been used successfully on programs such as AE, DE, ISEE, and numerous JPL deep space missions. The GSTDN and TDRSS transponder combination provides total redundancy. The shaped S-band omni antenna flew on ERBS and COBE and has not had any failures.

TIMED-H ELLIPTICAL ORBIT WORST CASE DOWNLINK ANALYSIS

Antenna Size	9 m	Frequency	2287.5 MHz
Transponder Power 5 W	5 W		
Altitude	6000.0 km		
Slant Range	10,286 km	Antenna Mask	5° Elevation

Parameter	Unit	Value
Transmitter Power	dBM	37.0
Spacecraft Passive Losses	dB	-2.5
Spacecraft Antenna Gain	dBI	0.0
Space Loss	dB	-179.9
Atmospheric Loss	dB	-0.3
Rain Attenuation	dB	
Polarization Loss	dB	-0.3
Ground Antenna Pointing Loss	dB	
Received Power (Isotropic)	dBM	-146.0
Ground Antenna G/T	dB	25.0
Carrier to Noise Density C/N_O	dB-Hz	77.6
Modulation Loss	dB	
Available E_b/N_O for a data rate 1500 kbps	dB	15.9
Required E_b/N_O for a BER = 1×10^{-6}	dB	11.6
Total Margin	dB	4.3

Table 5-15

TIMED-H CIRCULAR ORBIT WORST CASE DOWNLINK ANALYSIS

Antenna Size	9 m	Frequency	2287.5 MHz
Transponder Power	5 W	Altitude	400.0 km
Slant Range	1,851 km	Antenna Mask	5° Elevation

Parameter	Unit	Value
Transmitter Power	dBM	37.0
Spacecraft Passive Losses	dB	-2.5
Spacecraft Antenna Gain	dBI	0.0
Space Loss	dB	-165.0
Atmospheric Loss	dB	-0.3
Rain Attenuation	dB	
Polarization Loss	dB	-0.3
Ground Antenna Pointing Loss	dB	
Received Power (Isotropic)	dBM	-131.1
Ground Antenna G/T	dB	25.0
Carrier to Noise Density C/N_o	dB-Hz	92.5
Modulation Loss	dB	
Available E_b/N_o for a data rate 3000 kbps	dB	27.7
Required E_b/N_o for a BER = 1×10^{-6}	dB	11.6
Total Margin	dB	16.1

Table 5-16

TIMED-H ELLIPTICAL ORBIT WORST CASE UPLINK ANALYSIS

Antenna Size	9 m	Frequency	2106.4 MHz
Antenna Gain	44 dB	Altitude	6,000 km
Slant Range	10,286 km	Antenna Mask	5°Elevation
Transmitter	200 W	Data Rate	2.0 kbps

Command Mode Only - PCM/PSK/PM - 16 KHz Subcarrier

Parameter	Unit	Value
Effective Radiated Power	dBM	97.0
Space Loss	dB	-179.2
Atmospheric Loss	dB	-0.3
Polarization Loss	dB	-0.3
Spacecraft Antenna Gain	dBI	0.0
Spacecraft Passive Losses	dB	-2.5
Total Received Power	dBM	-85.2
Spacecraft Antenna Null Depth	dB	0.0
System Noise Density	dB-Hz	-169.3
IF Noise BW (3000.0 KHz)	dB-Hz	64.8
IF Noise Power	dBM	-104.5
IF SNR	dB	19.3
Required SNR	dB	0.0
Available IF Margin	dB	19.3
Required Performance Margin	dB	3.0
IF Net Margin	dB	16.3
Command Channel (PCM/PSK/PM)		
Total Power (MI = 0.50 RAD)	dB	-9.3
Received Command Power	dB	-94.5
Predetection (PSK) Noise Bandwidth (32.0 KHz)	dB-Hz	45.1
Predetection Noise Power	dB	-124.2
Predetection SNR	dB	29.7
Command Data Rate (2.0 kbps)	dB-bps	33.0
Available Energy Per Bit-Noise Density	dB	74.8
Decoder Degradation	dB	-2.0
Required Energy Per Bit-Noise Density (BER= 1×10^{-6})	dB	11.6
Available Command Margin	dB	61.2
Required Performance Margin	dB	3.0
Net Margin	dB	58.2

Table 5-17

TIMED-H AND TIMED-L TDRSS FORWARD LINK ANALYSIS

S-Band Single Access Forward Link Frequency = 2106.4 MHz Data Rate = 1.0 kbps Coherent Link		
Parameter	Unit	Value
1. TDRSS EIRP	dBW	43.5
2. Free Space Loss	dB	-191.9
3. Polarization Loss	dB	-0.5
4. User Antenna Gain	dB	0.0
5. User Antenna Pointing Loss	dB	-0.2
6. User Passive Loss	dB	-2.5
7. User Received Power	dB	-151.6
8. User Compatibility Loss	dB	0.0
9. Atmospheric Loss	dB	*
10. RFI Loss	dB	*
11. Dynamics Loss	dB	*
12. Antenna Switching Loss	dB	*
13. User Effective Received Power	dB	-151.6
14. User Noise Sensitivity	dBW-Hz	-201.5
15. User Received P/N_o	dB-Hz	49.9
16. User Required Acquisition P/N_o	dB-Hz	39.5
17. User Acquisition Margin	dB	10.4
18. Command to Total Power Ratio		-0.5
19. User Transponder Loss	dB	2.4
20. Received Command P/N_o	dB	47.0
21. Command Data Rate	dB-Hz	30.0
22. User Received E_b/N_o	dB	17.0
23. User Required E_b/N_o	dB	10.8
24. Effective User Command Margin	dB	6.2

Table 5-18

TIMED-H AND TIMED-L TDRSS RETURN LINK ANALYSIS

S-band Single Access
Transponder Power 5 W
Return Link Frequency 2287.5 MHz
DG 1/Mode 1

I Channel	Q Channel
Data Rate = 1 kbps	Data Rate = 1 kbps
Mod 2 Added to PN	Mod 2 Added to PN
Data Type = NRZ-L	Data Type = NRZ-L
Rate 1/2 Coded	Rate 1/2 Coded

Parameter	Unit	I Channel	Q Channel
User Transmitter Power	dB	7.00	7.00
User Passive Loss	dB	-2.50	-2.50
User Antenna Gain	dB	0.00	0.00
User Pointing Loss	dB	0.00	0.00
Polarization Loss	dB	0.00	0.00
User Data/Total Power Ratio	dB	-3.00	-3.00
Free Space Loss	dB	-192.61	-192.61
TDRSS Received Power-Unity Gain	dB	-191.11	-191.11
User Constraint Loss	dB	0.00	0.00
Other Losses	dB	*	*
RFI Environment Loss	dB	-0.50	-0.50
Degraded Effective Power Unit	dB	-191.61	-191.61
Required Effective Power Unit	dB	-196.90	-196.90
Effective User Margin	dB	5.29	5.29
Combined User Margin	dB	10.60	

Table 5-19

TIMED-H ORBITAL CONTACT TIME ANALYSIS

Date	View in Minutes	Remarks
02-01-99	77	
02-12-99	54	
		Perigee Alt. = 132
03-03-99	32	Perigee at 85° N
03-24-99	51	
04-06-99	65	
04-30-99	128	
05-25-99	192	
06-01-99	204	Perigee at 85° S
06-15-99	197	
06-25-99	174	
		Perigee Alt. = 160
07-01-99	152	
07-05-99	142	
07-23-99	73	
08-10-99	35	
08-27-99	20	Perigee at 85° N
09-10-99	36	
09-27-99	62	
10-14-99	92	
10-21-99	101	
10-31-99	104	
12-21-99	15	Apogee Decays
12-24-99	14	Perigee at 132
12-27-99	12	Perigee at 85° N
12-30-99	10	
01-02-00	14	
01-11-00	18	
01-17-00	18	
01-20-00	17	
01-23-00	14	
		Circular Orbit
01-23-00	21	128 Right Asc.
01-24-00	18	
01-25-00	17	
01-26-00	22	
01-27-00	14	
01-28-00	21	
01-29-00	14	
01-30-00	22	
01-31-00	18	
02-01-00	17	

Date	View in Minutes	Remarks
06-05-00	17	218 Right Asc.
06-06-00	19	
06-07-00	19	
06-08-00	15	
06-09-00	21	
06-10-00	13	
06-11-00	21	
06-12-00	16	
06-13-00	20	
06-14-00	20	
10-18-00	15	308 Right Asc.
10-19-00	19	
10-20-00	18	
10-21-00	16	
10-22-00	20	
10-23-00	13	
10-24-00	22	
10-25-00	13	
10-26-00	21	
10-27-00	19	
03-02-01	21	38 Right Asc.
03-03-01	16	
03-04-01	21	
03-05-01	14	
03-06-01	22	
03-07-01	15	
03-08-01	19	
03-09-01	19	
03-10-01	15	
03-11-01	22	
07-02-01	21	119 Right Asc.
07-03-01	18	
07-04-01	17	
07-05-01	18	
07-06-01	17	
07-07-01	20	
07-08-01	15	
07-09-01	22	
07-10-01	18	
07-11-01	18	

Table 5-20

TIMED-L ORBITAL CONTACT TIME ANALYSIS

Date	View In Minutes	Remarks
02-01-99	43	311 Right
02-02-99	42	Asc.
02-03-99	34	
02-04-99	40	
02-05-99	37	
02-06-99	40	
02-07-99	34	
02-08-99	42	
02-09-99	42	
02-10-99	32	
02-11-99	43	
02-12-99	38	
02-13-99	40	
02-14-99	46	
04-10-99	34	
04-11-99	41	310 Right
04-12-99	35	Asc.
04-13-99	43	
04-14-99	32	
04-15-99	42	
04-16-99	36	
04-17-99	37	
04-18-99	37	
04-19-99	37	
04-20-99	34	
04-21-99	37	
04-22-99	37	
04-23-99	35	
06-01-99	37	
06-02-99	38	
06-03-99	37	38 Right
06-04-99	37	Asc.
06-05-99	37	
06-06-99	38	
06-07-99	36	
06-08-99	36	
06-09-99	37	
06-10-99	34	
06-11-99	37	
06-12-99	37	
06-13-99	37	
06-14-99	37	

Date	View In Minutes	Remarks
08-01-99	35	72 Right
08-02-99	45	Asc.
08-03-99	35	
08-04-99	40	
08-05-99	37	
08-06-99	41	
08-07-99	38	
08-08-99	36	
08-09-99	43	
08-10-99	32	
08-11-99	43	
08-12-99	35	
08-13-99	40	
08-14-99	41	
09-30-99	37	
10-01-99	37	112 Right
10-02-99	37	Asc.
10-03-99	35	
10-04-99	38	
10-05-99	37	
10-06-99	37	
10-07-99	37	
10-08-99	37	
10-09-99	36	
10-10-99	37	
10-11-99	36	
10-12-99	36	
10-13-99	37	
12-01-99	44	
12-02-99	36	
12-03-99	38	145 Right
12-04-99	33	Asc.
12-05-99	43	
12-06-99	38	
12-07-99	35	
12-08-99	44	
12-09-99	35	
12-10-99	41	
12-11-99	35	
12-12-99	43	
12-13-99	39	
12-14-99	36	

Table 5-21

TIMED RF COMMUNICATIONS SUBSYSTEM FAILURE EFFECTS ANALYSIS

Failure	Backup Mode	Mission Impact
GSTDN Transponder	Use Redundant TDRSS Transponder	Degraded System Performance due to increased power required
TDRSS Transponder	Use GSTDN Transponder	Degraded Performance Reduced Attitude Knowledge because ranging contacts limited to WOTS
RF Switch	Use Failed Position Antenna	Degraded Performance Shorter Contact Times
S-Band OMNI Antenna	Use Redundant Antenna	Degraded Performance Shorter Contact Times

Table 5-22

5.10 Electrical Subsystem

5.10.1 Requirements

The electrical subsystem satisfies the following requirements:

- Provide the electrical interconnection of all subsystem and instrument components.
- Distribute primary observatory power and signals to all users. Signal distribution shall minimize common impedances and maximize signal isolation within the observatory.
- Provide -28 ±6 VDC input to subsystems and instruments.
- Provide system grounding, including structural and electrical grounding.
- Meet harness wire derating in accordance with the GSFC Preferred Parts List.
- Incorporate wire and connector contact redundancy for all circuits which are considered to be critical.
- Comply with EMI/EMC requirements.
- Implement point-to-point connections of MIL-STD-1773 fiber optic bus.

5.10.2 Subsystem Description

The electrical subsystem consists of all observatory harnesses, grounding straps for boxes and thermal blankets, cable shields and miscellaneous hardware to secure the harness to the structure.

The electrical subsystem distributes power to all spacecraft loads via three major electrical buses: essential, nonessential, and pulse. The essential bus supports critical subsystem and instrument circuits. The nonessential bus provides switched power to non-essential spacecraft loads. Pulse loads, such as the EED's, are those which require single intermittent power pulses for operation.

The harness design is modular, consisting of three parts; one section for the instrument mounting plate, one for the propulsion module, and one for the subsystems with appropriate interconnections. The harness design also minimizes stray magnetic fields by twisting pairs of wires to achieve a net current in each cable.

The harness, grounding, bonding, and shielding scheme is specified in the TIMED GIIS to ensure compliance with EMI/EMC requirements. A positive potential, modified single point, electrical grounding scheme is represented in Figure 5-43. The primary power return is conductively grounded to the spacecraft structure at one point only, preferably at or near the Power Supply Electronics.

All conductive exterior surfaces, including thermal blankets and spacecraft close-outs, but exclusive of intentionally biased instrument and subsystem probes and guards, shall be grounded to the observatory structure. No electrical conductors that have voltages other than that of the spacecraft structure shall be directly exposed to the external environment. All exterior wiring and connections, such as solar cell conductive tabs, must be completely insulated.

Primary power is isolated from the spacecraft structure and electrical circuits by at least 10 megohms resistance and shunted by no more than 10 nanofarads capacitance. Secondary power

return is referenced to signal ground at the load and is connected to the chassis. Each component is electrically bonded to the spacecraft structure with a resistance not exceeding 2.5 milliohms.

5.10.3 Performance and Analysis

The electrical subsystem meets all system requirements.

5.10.4 Reliability, Heritage and Risk Assessment

System reliability is achieved by implementation of the design principles and requirements developed over many years at GSFC.

Fiber optic components will have been flown on XTE and TRMM. Radiation induced problems have been shown to be negligible based on in-orbit performance of similar optical components on the Long Duration Exposure Facility (LDEF).

Compliance with EMI/EMC requirements is demonstrated by implementation of standard global specifications; i.e., GEVS-SE and MIL-STD-461/462.

The electrical subsystem design and requirements implementation are typical of GSFC missions. Risk is considered negligible.

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TIMED SINGLE POINT GROUND SCHEME

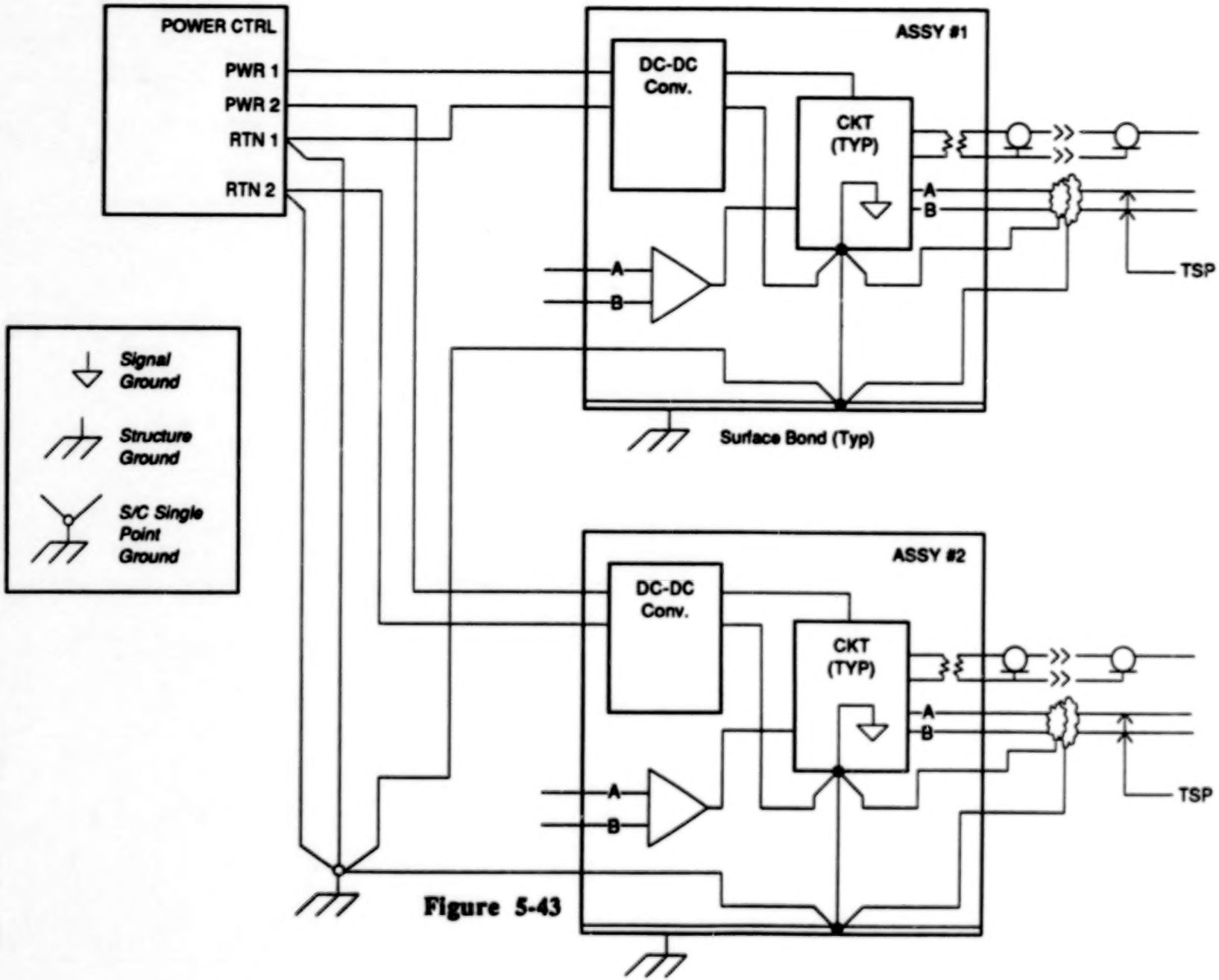


Figure 5-43

5.11 Observatory Mass, Power, and Data Rate Summary

Table 5-23 provides a summary of mass, power and data rates for both the TIMED-H and TIMED-L observatories. The launch vehicle capability is based on three Delta II solid rocket boosters. Table 5-24 provides a further resource breakdown of the spacecraft subsystems.

TIMED OBSERVATORY MASS, POWER AND DATA RATE SUMMARY

	Mass (kg)		Power (W)		Data (kbps)	
	TIMED-H	TIMED-L	TIMED-H	TIMED-L	TIMED-H	TIMED-L
Launch Vehicle Capability	1118.0	2491.0	N/A	N/A	N/A	N/A
Observatory Total	945.71	882.97	214.77	283.11	27.52	15.53
Propellant	215.40	113.70	N/A	N/A	N/A	N/A
Margin	0.00	0.00	43.00	56.67	5.50	3.11
Sub-Total	730.31	769.27	171.77	226.44	22.02	12.43
Instr. Duty Cycle	N/A	N/A	0.25	0.25	1.0	1.00
Instruments @%D.C.	90.80	112.50	25.75	29.50	19.02	10.30
IRLS Standby	N/A	N/A	0.00	50.00	TBD	TBD
ACS	77.38	68.38	44.49	44.56	0.30	0.30
C&DH	33.80	32.80	39.92	38.77	0.30	0.30
Communications	22.65	22.65	8.90	8.90	0.15	0.15
Electrical	54.10	56.98	6.61	8.71	0.00	0.00
Power	112.49	119.62	10.00	10.00	0.10	0.10
Propulsion	66.66	66.66	1.00	1.00	0.05	0.05
Structure	243.33	260.18	0.00	0.00	0.00	0.00
Thermal	29.10	29.50	35.00	35.00	0.10	0.10
Science Instruments	90.80	112.50	103.00	168.00	19.016	10.296
Fabry-Perot	9.60	9.60	7.00	7.00	1.28	1.28
UV Spectrometer	7.30	0.00	5.00	0.00	0.06	0.00
Imag. Photometer	8.40	8.40	10.00	10.00	1.28	1.28
Near IR Spect.	5.00	0.00	7.50	0.00	0.26	0.00
UV Airglow Imag.	20.00	0.00	30.00	0.00	11.00	0.00
N. Mass Spect.	12.00	12.00	24.00	24.00	1.54	1.54
Ion Drift Meter	8.00	8.00	8.00	8.00	2.20	2.20
Langmuir Probe	2.50	2.50	4.00	4.00	1.00	1.00
Solar EUV Spect.	18.00	0.00	7.50	0.00	0.40	0.00
IR Limb Sounder	0.00	72.00	0.00	65.00	0.00	3.00
IRLS Standby	0.00	0.00	0.00	50.00	0.00	0.00

Table 5-23

TIMED SPACECRAFT SUBSYSTEM MASS, POWER AND DATA RATES

Subsystem	Mass (kg)		Power (W)		Data (kbps)	
	TIMED-H	TIMED-L	TIMED-H	TIMED-L	TIMED-H	TIMED-L
ACS	77.38	68.38	44.59	44.56	0.30	0.30
Type	Bias Mom	Bias Mom	Bias Mom	Bias Mom		
Earth Sensor-1	2.60	2.60	4.00	4.00		
Earth Sensor-2	2.60	2.60	4.00	4.00		
Sol. Asp. Sensor (2)	1.60	1.60	2.00	2.00		
Triax Magnetometer	1.20	1.20	3.00	3.00		
3 Torquerods	21.78	21.78	17.88	17.88		
Moment Wheel-1	14.30	14.30	2.84	2.84		
Moment Wheel-2	14.30	14.30	2.84	2.84		
C.G. Comp.-1	3.00	0.00	0.01	0.00		
C.G. Comp.-2	3.00	0.00	0.01	0.00		
C.G. Comp.-3	3.00	0.00	0.01	0.00		
Nutation Damper	0.00	0.00	0.00	0.00		
ACS Electronics	10.00	10.00	8.00	8.00		
C&DH	33.80	32.80	39.92	38.77	0.30	0.30
C&DH Unit	27.80	26.80	36.92	35.77	0.30	0.30
Freq. STD-1	1.00	1.00	3.00	3.00		
Freq. STD-2	1.00	1.00	0.00	0.00		
Star Couplers	4.00	4.00	0.00	0.00		
Peak-Traffic (kbps)					27.52	15.53
Margined Rate (kbps)					27.52	15.53
Storage Needed (Gb)					2.38	1.34
Storage Provided					2.68	2.01
Communications	22.65	22.65	8.90	8.90	0.15	0.15
Orbital Conditions			400 Circ.	400 Circ.		
TDRSS Xpndr@	6.33	6.33	4.90	4.90	0.00	0.00
Xmit Minutes/Day			156	156		
GSTDN Xpndr@	3.50	3.50	4.00	4.00	0.00	0.00
Xmit Minutes/Day			13	13		

Table 5-24

TIMED SPACECRAFT SUBSYSTEM MASS, POWER AND DATA RATES (CONTD.)

Subsystem	Mass (kg)		Power (W)		Data (kbps)	
Communications	**TIMED-H**	**TIMED-L**	**TIMED-H**	**TIMED-L**	**TIMED-H**	**TIMED-L**
Antenna-1	4.54	4.54	0.00	0.00	0.00	0.00
Antenna-2	4.54	4.54	0.00	0.00	0.00	0.00
Filter-1	0.57	0.57	0.00	0.00	0.00	0.00
Filter-2	0.57	0.57	0.00	0.00	0.00	0.00
Diplexer-1	0.60	0.60	0.00	0.00	0.00	0.00
Diplexer-2	0.60	0.60	0.00	0.00	0.00	0.00
Misc. H/W & Cabling	1.40	1.40	0.00	0.00	0.00	0.00
Propulsion	66.66	66.66	1.00	1.00	0.05	0.05
8 Thrusters	4.02	4.02	0.00	0.00	0.05	0.05
6 MD Tanks	44.90	44.90	0.00	0.00		
Manifold	5.34	5.34	0.00	0.00		
10 Valves	2.72	2.72	0.00	0.00		
2 Filters	0.28	0.28	0.00	0.00		
Press Xducers	0.58	0.58	1.00	1.00		
Misc. Brkts.	8.82	8.82	0.00	0.00		
Electrical	54.10	56.98	6.61	8.71	0.00	0.00
% of Total	8.00	8.00	4.00	4.00		
Power	112.49	119.62	10.00	10.00	0.10	0.10
Pwr. Sply. Elect.	25.00	25.00	10.00	10.00		
20AH NcCd Batt.	21.00	21.00	0.00	0.00		
20AH NcCd Batt	21.00	21.00	0.00	0.00		
Shunts	2.00	2.00	0.00	0.00		
S/A Body	3.00	3.00	0.00	0.00		
S/A Deploy	35.50	35.50	0.00	0.00		
S/A End	0.00	7.13	0.00	0.00		
	5.00	5.00	0.00	0.00		
S/A Boxy (Sq M)	10.66	10.66			0.30	0.30
S/A Deploy	0.00	2.01				
S/A End (Sq M)	1.50	1.50				
KG/(Sq M) Bdy. & End	3.33	3.33				
KG/(Sq M) Deploy	3.51	3.51				

Table 5-24

TIMED SPACECRAFT SUBSYSTEM MASS, POWER AND DATA RATES (CONTD.)

Subsystem	Mass (kg)		Power (W)		Data (kbps)	
	TIMED-H	TIMED-L	TIMED-H	TIMED-L	TIMED-H	TIMED-L
Structure	243.33	260.18	0.00	0.00	0.00	0.00
S/A Substrates	113.33	124.98	0.00	0.00		
Balance Mass	6.50	6.50	0.00	0.00		
Bottom Deck	19.50	19.50	0.00	0.00		
Central Frame	31.20	31.20	0.00	0.00		
Shear Panels	5.20	5.20	0.00	0.00		
Prop. Sup. Struct.	14.30	14.30	0.00	0.00		
Misc. H/W	13.00	13.00	0.00	0.00		
Top Deck	35.10	35.10	0.00	0.00		
Top Deck Frame	5.20	5.20	0.00	0.00		
Booms/Mech.	0.00	5.20	0.00	0.00		
(Adapter Ftng.)	57.00	57.00	0.00	0.00		
Thermal	29.10	29.50	35.00	35.00	0.10	0.10
Mult. Layr. Insul.	8.00	8.00	0.00	0.00		
Heat Pipes	14.00	14.40	0.00	0.00		
Heat Cntlrs.	0.50	0.50	10.00	10.00		
Heaters	2.00	2.00	25.00	25.00		
Pain	0.30	0.30	0.00	0.00		
Louvers	4.30	4.30	0.00	0.00		

Table 5-24

5.12 Summary of Phase A Study Options

This section describes the nine options considered during the TIMED Phase A Study. Six of these options were developed by the Science Definition Team (SDT) in response to a NASA Headquarters' mandate to accomplish this program within a $300M budget. The SDT developed the six options by varying the instrument complement, the number of spacecraft, the mission duration, and the orbital profiles. The science rationale for the option definition is described further in Section 3.5. One additional option was developed by the project team to investigate the potential savings associated with using five small explorer class spacecraft buses to perform the TIMED Mission. Two additional options were defined by the study team and the study scientist (working in close coordination with SDT members) as variants off of Option 1 and 2 (denoted as 1' and 2').

The project team developed spacecraft conceptual designs per the option requirements and these designs were costed by the Resource Analysis Office (RAO). A Phase A concept was down-selected from this set of options. Additional analysis and design work was performed on this down-selected option to produce the Phase A Study conceptual design described in this report.

This section is divided into three topics: 1) Section 5.12.1 presents an overview of the nine options; 2) Section 5.12.2 describes the spacecraft subsystems; and 3) Section 5.12.3 provides the spacecraft design details for each option.

5.12.1 Option Descriptions

The instrument complement for each option in presented in Table 5-25. The major mission requirements for each option is summarized in Table 5-26.

Figure 5-44 presents the orbit profiles for each option. All options, except Option 5, require spacecraft in a high (TIMED-H) inclination and in a low (TIMED-L) inclination orbit. Option 5 has only one observatory in a compromise orbital inclination of 70 degrees. Options 0, 1, 1C, and 2 have an orbital inclination of 95 degrees for TIMED-H and 49 degrees for TIMED-L. Option 4 requires orbital inclinations of 70 degrees for TIMED-H and 30 degrees for TIMED-L.

The TIMED-H spacecraft is in an elliptical orbit at the beginning of the mission to take advantage of the launch vehicle capability to insert the spacecraft into a high energy orbit. Except for Options 1' and 2', the TIMED-L observatory is initially circular and is maneuvered into an elliptical orbit 18 to 24 months from mission start. The minimum perigee altitude is dependent on the option.

Table 5-26 also identifies the key changes made by the SDT as they moved down the option list towards less science and reduced program costs. A summary of these changes is presented below:

A. The differences between Option 0 and 1 are the mission lifetime and instrument duty cycle. Mission lifetime was reduced from 4 to 2 years with the requirement to carry sufficient propellant to maintain the orbit for 4 years. Instrument duty cycle is reduced from 100 percent to 25 percent.

B. The next step reduces the number of instruments from 23 to 15. This reduces the instrument complement cost and the resources that the spacecraft has to supply to these instruments.

INSTRUMENT COMPLEMENT FOR OPTIONS 0-5

INSTRUMENT	Option 0/1		Option 2		Option 3		Option 4		Option 5
	H	L	H	L	H	L	H	L SMEX	~70° Incl.
Fabry-Perot Interferometer	X	X	X	X	X	X	X	X	X
UV Spectrometer	X	X	X		X		X	X	X
Imaging Photometer	X	X	X	X	X	X	X	X	X
Near Infrared Spectrometer/Photometer	X		X		X		X		X
Global UV Airglow Imager	X		X		X		X		X
Neutral Mass Spectrometer/ Wind and Temperature	X	X	X	X	X	X	X		X
Ion Mass Spectrometer	X	X							X
Ion Drift Meter/Retarding Potential Analyzer	X	X	X	X	X	X	X		X
Langmuir Probe	X	X	X	X	X	X	X		X
Energetic Particles Analyzer	X								X
Vector Magnetometer	X								X
Solar EUV Spectrometer	X		X		X		X		X
Satellite Electrostatic Triaxial Accelerometer	X	X							X
Electric Field Detector/Plasma Wave Experiment	X								X
Infrared Limb Sounder		X		X		X	X		X
TOTALS	14	9	9	6	9	6	10	3	15

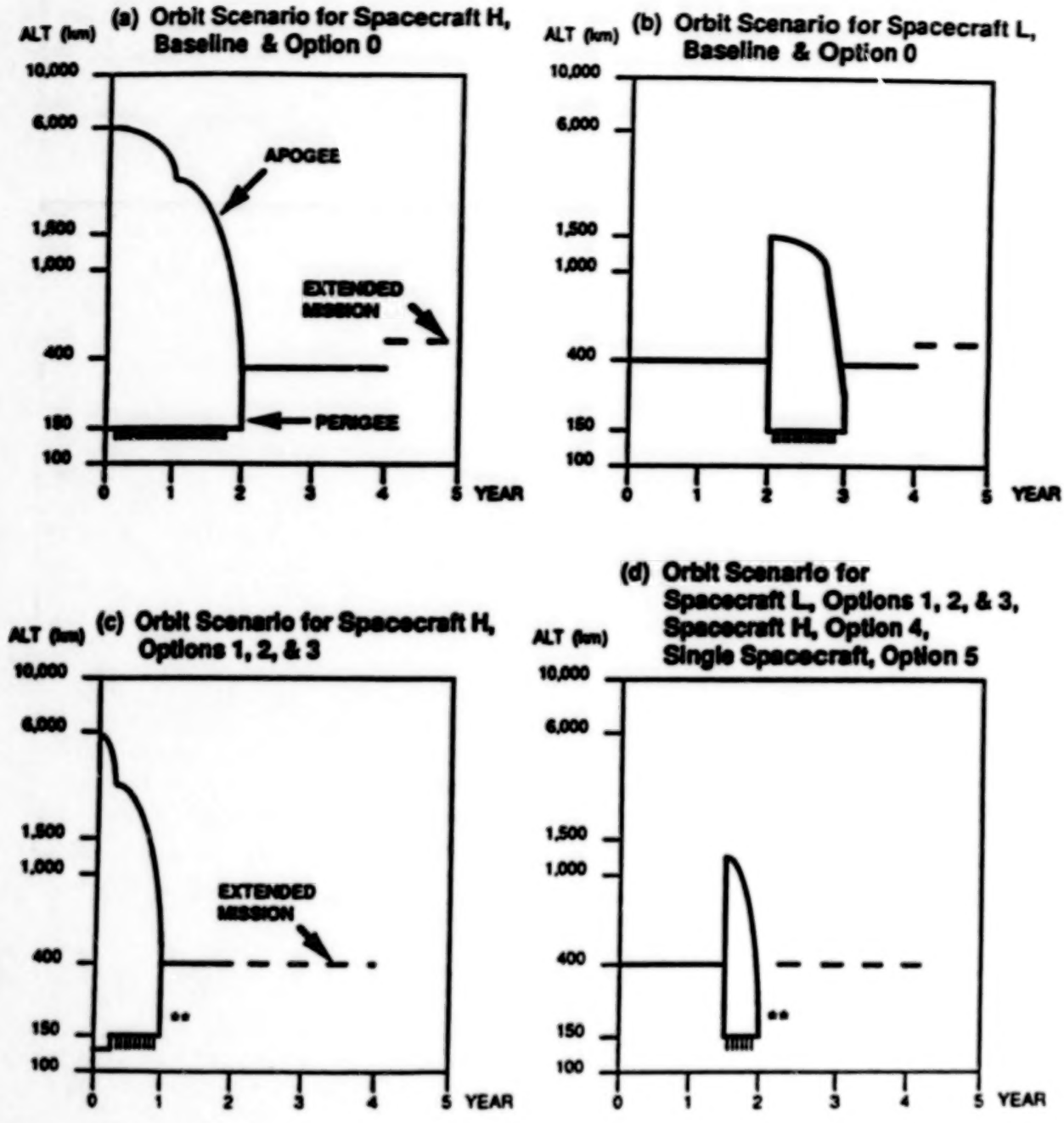
Table 5-25

OPTION OVERVIEW

FROM SCIENCE DEFINITION TEAM	
OPTION 0:	4 YEAR MISSION, 2 SPACECRAFT *(23 INSTRUMENTS/15 UNIQUE)*
OPTION 1:	REDUCED MISSION TO 2 YEARS *(23 INSTRUMENTS/15 UNIQUE)*
OPTION 2:	REMOVED 8 INSTRUMENTS INCLUDING BOOM INSTRUMENTS *(15 INSTRUMENTS/10 UNIQUE)*
OPTION 3:	REMOVED REQUIREMENT FOR DEEP DIPPING *(15 INSTRUMENTS/10 UNIQUE)*
OPTION 4:	LARGE SPACECRAFT AT 70° INCLINATION *(10 INSTRUMENTS/10 UNIQUE)* SECOND SPACECRAFT "SMALL EXPLORER AT 30° INCLINATION *(3 INSTRUMENTS/NO UNIQUE)*
OPTION 5:	ONE SPACECRAFT AT 70° INCLINATION - INITIALLY CIRCULAR WITH DEEP DIPPING PHASE *(15 INSTRUMENTS/15 UNIQUE)*
FROM PHASE A STUDY TEAM	
OPTION 1C:	SAME AS OPTION 1 EXCEPT USES 5 SMALL "SMEX CLASS" SATELLITES
OPTION 1':	SAME AS OPTION 1 WITH ELIMINATION OF ELLIPTICAL PHASE ON TIMED-L
OPTION 2':	SAME AS OPTION 2 WITH ELIMINATION OF ELLIPTICAL PHASE ON TIMED-L

Table 5-26

TIMED ORBIT PROFILES FOR THE PROPOSED OPTIONS



** Perigee and dip altitudes are adjusted as indicated in option descriptions.

Figure 5-44

C. The next parameter to be relaxed is the minimum perigee altitude. By eliminating the requirement for the spacecraft to "deep dip" down to 130 km altitude, the complexity and cost of the spacecraft attitude control and propulsion subsystems are reduced. The relaxation of this parameter also allows the use of deployed solar panels and simplified thermal subsystem design.

D. The next step reduced the number of instruments from 15 to 13 and introduced the use of a small spacecraft (small explorer class) as one of the TIMED spacecraft. As a result of redistributing the instruments to accommodate the small spacecrafts' capabilities, the orbital inclinations were adjusted to 70 degrees and 30 degrees for TIMED-H and TIMED-L respectively.

E. The final SDT reduction eliminated one spacecraft. To minimize the loss of science, additional instruments are added to increase the number from 13 to 15 and the low minimum perigee altitude requirement is reinstated.

The project team created 3 options in addition to the 6 options documented by the SDT. These are Options 1C, 1', and 2'. The Options 1' and 2' were developed in close coordination with the study scientist.

Option 1C uses 5 "small explorer" (SMEX) class spacecraft to fly all the instruments from Option 1. For this reason, it was considered by the engineers to have the same science value as Option 1. The thinking behind this option was, that if a common spacecraft bus design could be used for all 5, it would be a quicker, smaller and cheaper way of getting all the science for Option 1. Even though there are 5 spacecraft, the smaller launch vehicle would result in lower overall cost to the agency. The "1C" nomenclature is retained to provide traceability to other study documentation. Option 1B was renumbered to be Option 2.

Option 1' eliminates the elliptical phase on TIMED-L from Option 1 with the observatory in a circular orbit at 400 km for the entire mission. The reboost required to take TIMED-L from a circular orbit to an elliptical orbit, with apogee at 1500 km, turned out to be a significant driver to the size of the propulsion system and, consequently, the size, weight and cost of the observatory. Since the two observatories have the same design, it drove the size, weight and cost of both observatories.

Option 2' is identical to Option 2, except that the elliptical phase is eliminated on TIMED-L, as in the description of Option 1' above.

5.12.2 Spacecraft Subsystem Conceptual Design

A unique spacecraft configuration was developed to support the options' instrument complement and orbital profiles. Eight spacecraft subsystems were defined to meet the option requirements and their components were characterized as to mass, power, volume, and data requirements.

Command and data handling (C&DH) and RF communications are nearly common across all options. The only C&DH subsystem parameter that was adjusted is the small change in weight and power to adjust the solid state recorder size to meet instrument data requirements. For the RF communications subsystem, an analysis was performed for all the options to verify that 24 hours worth of stored science data could be retrieved from each satellite via the Wallops Orbital Tracking Station.

The electrical subsystem harness design is defined as a percent of the spacecraft dry weight and power, based on historical data. The harness mass is 8 percent of the total spacecraft dry mass, and the electrical subsystem power requirement is sized at 4 percent of the total observatory power.

The spacecraft subsystem design process for the other subsystems consisted of the following steps. The low minimum perigee altitude requirement, coupled with instrument viewing and calibration requirements and efficient power generation geometry, resulted in a cylindrically shaped spacecraft for most of the options. The first step in the design process sized the solar arrays for the 25 percent instrument duty cycle in the defined orbits. The array size translates to a body size from which the orbital drag is calculated and this determines the fuel required to meet the mission profile and lifetime. Once sized and fueled, the spacecraft attitude control subsystem (ACS) performance is evaluated for pointing errors as a function of momentum wheel capabilities. Structural mass is then updated for a refined mass sizing and fuel calculation. The thermal subsystem then designs its components to maintain the solar arrays, instruments and spacecraft subsystems within nominal temperature boundaries. This iterative process is repeated at least two times during the development of each option spacecraft.

Some options provide for additional relaxation of the requirements and different subsystem configurations were developed to investigate their impact on program costs.

Each of the following spacecraft subsystems is fully described earlier in Section 5.0; therefore, this subsystem only describes variations from that basic subsystem.

ACS - All options, except Option 4, utilize a biased momentum system. Within each option, the size of the momentum wheel is adjusted to control the aerodynamic torque resulting from the cg/cp offset. The magnetic torque rods are sized to unload the momentum buildup during the low perigee altitude portion of the orbit on a per orbit basis. For each iteration, all components except the momentum wheels and magnetic torque rods remain fixed. Those two components are adjusted to allow the ACS to meet the spacecraft attitude control requirements. The Option 4 ACS is 3-axis stabilized to investigate the cost and performance impact of this type of system in a spacecraft not required to traverse the low minimum perigee altitude.

Propulsion - To provide a coarse center-of-gravity/center-of-pressure (cg/cp) offset control mechanism, all of the low minimum perigee altitude options contain six propellant tanks so selective propellant use adjusts the cg location. Among the options, the tank size is the only component that changed. The propellant tanks are sized to accommodate the maximum fuel requirement between the two spacecraft. Fuel tank sizing is driven by the TIMED-L spacecraft because of the fuel required to boost the spacecraft from a circular to an elliptical orbit. Options 1' and 2' fuel requirements are determined by the less demanding TIMED-H orbit profiles.

Thermal - The use of one spacecraft end as a dedicated radiator requires a yaw sun-avoidance maneuver to be periodically performed for all options except 1C, 3 and 4. The subsystem components that varied among the options are radiator size and the number of heat pipes.

Power - This subsystem is the major driver in the conceptual designs. Solar arrays are sized based on mission life, the power requirements, and the orbits. GaAs cells are baselined for all the cylindrical configurations except Option 0. Silicon cells are used for all deployed solar array configurations and for Option 0.

Structure - The structure design evolved into a modular concept for all of the cylindrical spacecraft. For each option, the structure size and mass are changed to accommodate the solar array size, propellant tank size, and the spacecraft subsystems and instrument volume and layout requirements.

5.12.3 Option Conceptual Designs

The following paragraphs contain more detail on the conceptual design of each option.

Option 0

This option defines the upper limit of what could be provided in spacecraft design to maximize the science return. This option was finished prior to the finalization of the design guidelines and no subsystem performance margins or redundancy are included. The power subsystem is sized to provide a 100 percent instrument duty cycle and uses silicon cells. The propulsion system is sized at 30 percent of the dry weight. The design is driven by the TIMED-L spacecraft power requirement for 6 m^2 of solar array at the 4 year end of life. Both observatories are cylinders 2.3 m in diameter and 2.7 m long, with 18 sides, and are clad with solar cells except for instrument and subsystem aperture cutouts. The design for this option is shown in Figure 5-45.

Preliminary analysis indicated that the propulsion and ACS subsystems would require considerable upsizing to accommodate the large spacecraft size. Based on these concerns and the high cost figures, this option was not studied any further.

OPTIONS 0 AND 1 DESIGN CONCEPT

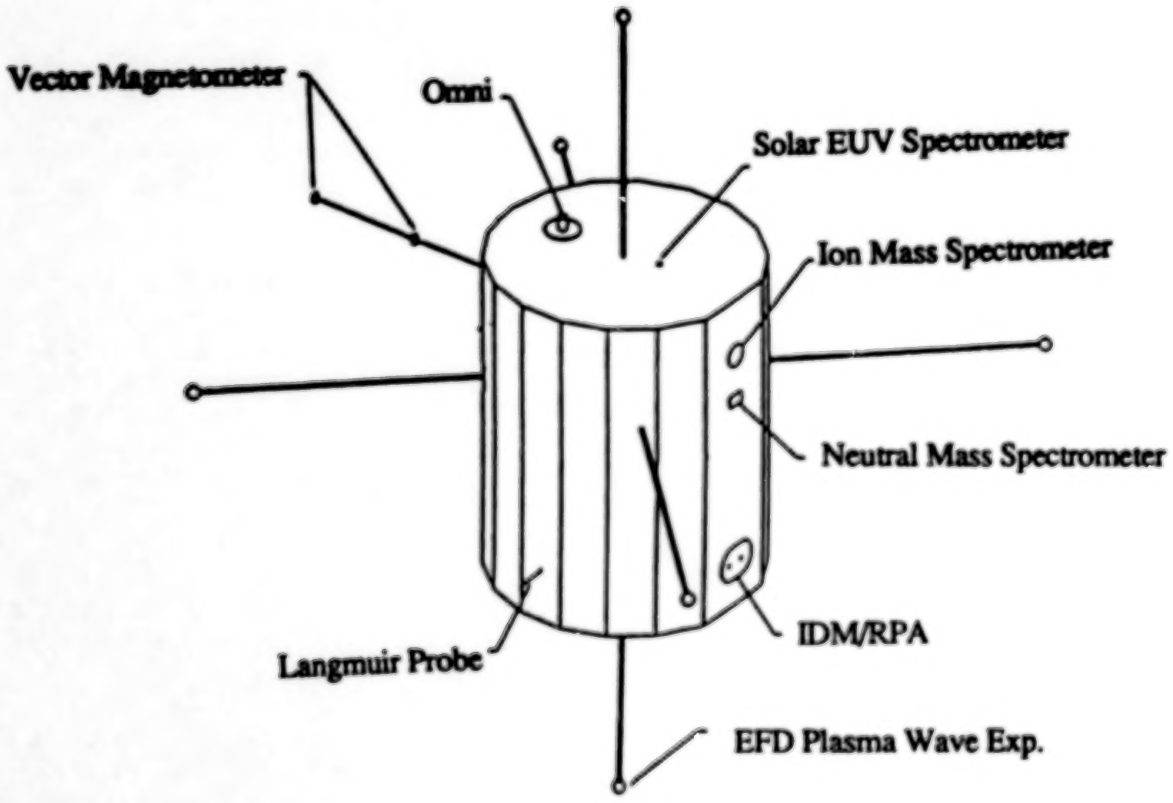


Figure 5-45

Option 1

This option reduces the mission lifetime to 2 years with the requirement for sufficient fuel to maintain the orbit for 4 years. The Study Team instituted design guidelines requiring subsystem performance margins and selected redundancy.

The spacecraft is an 18-sided cylinder with a 2.0 m diameter and 2.3 m length. The spacecraft concept is similar to the Option 0 spacecraft presented in Figure 5-45. The size is reduced by limiting the instruments to a 25 percent duty cycle and by using the more efficient and radiation resistant GaAs solar cells. The propulsion subsystem is sized to provide sufficient fuel to achieve the orbital profiles, allowing for 2 sigma atmospheric density variations and an 8 percent drag increase due to the boom instruments. The ACS subsystem is momentum bias with tight pitch control, but allows a 7 degree roll error during the low minimum perigee altitude orbits.

Option 1'

This option is identical to Option 1 except that the TIMED-L orbit is circular at 400 km altitude. This compromise between observatory design and science return allows a 316 kg reduction in the required fuel for the common spacecraft design. This fuel reduction allows additional reductions in spacecraft mass of 126 kg and the associated ACS downsizing reduces the power required by 69 watts.

Option 1C

This option study verified that the instruments could be physically placed onto five off-the-shelf small spacecraft as shown in Figure 5-46. However, these spacecraft would require significant redesign to provide adequate body-mounted solar array area and a more robust ACS for the low minimum perigee altitude phases. The radiation exposure from the Van Allen Belts during the elliptical orbit phases, and the design lifetime of one to two years, are concerns with respect to the spacecraft subsystem reliability. Some of the spacecraft would require the addition of a propulsion subsystem. Other subsystem upgrades include the ACS for pointing knowledge and the C&DH for enhanced telemetry data rates.

OPTION 1C CONCEPT USING FIVE SMALL SATELLITES

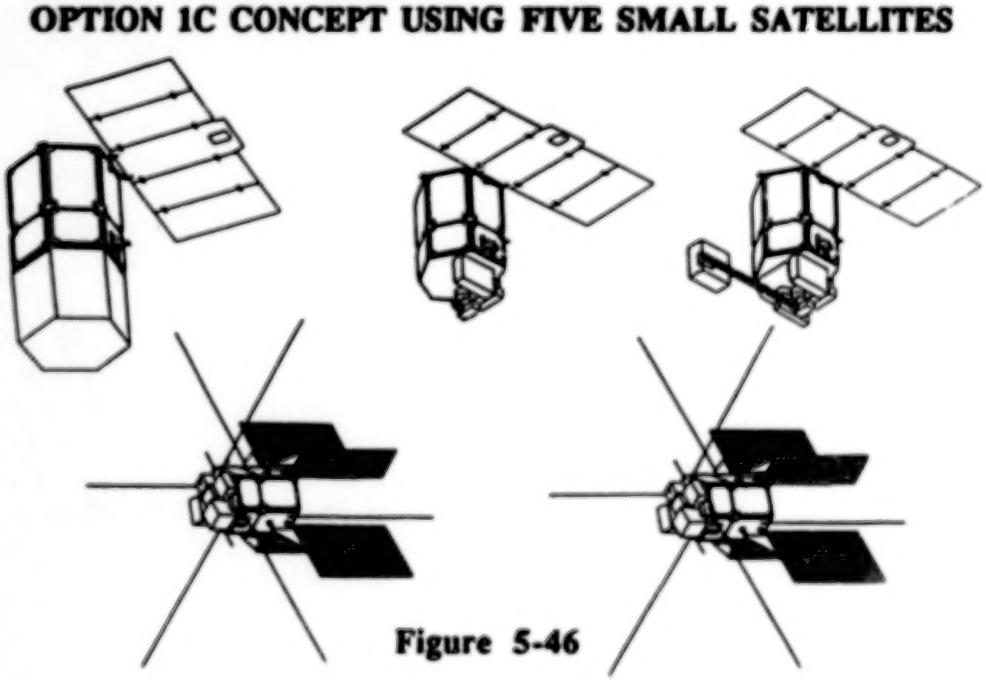


Figure 5-46

Additional concerns include, the required use of an unproven launch system, such as the TAURUS, for insertion into the elliptical orbits; the loss of concurrent science observations; the difficulty of launching five observatories in a short timeframe; and the additional burden of operating five observatories.

Because of the above considerations and the RAO cost figures, this option was not carried beyond the initial study stage.

Option 2

This option reduces the number of instruments from 23 to 15 for the two observatories. This did not result in any appreciable savings to the spacecraft design, except for the removal of all boomed instruments. The differences were small enough that the Option 2 conceptual design was defined to be identical to the Option 1 spacecraft.

Option 2'

This option is identical to Option 2 except that the TIMED-L orbit is circular at 400 km. Like Option 1', this change to the orbit profile reduces the Option 2 fuel by 316 kg, total observatory mass by 113 kg and power by 61 watts. These savings, plus the reduced cost for instrument development, integration and test resulted in a total mission cost close to the mandated $300M. As a result, this option was selected for additional analysis and design as the TIMED Phase A conceptual design.

Option 3

The introduction of a significantly higher minimum perigee altitude allows the use of deployed solar arrays. The project team developed a modular design that uses two, fixed, symmetrical silicon solar arrays, a triangular shaped body for the spacecraft subsystems and a box shaped housing for the science instruments. Figure 5-47 provides a description of this conceptual design. A total of 10.5 m^2 of solar array area is needed because of the fixed arrays and the cosine law inefficiency. Fixed solar arrays were selected over gimbaled, sun-tracking arrays to avoid field-of-view conflicts and to reduce cost and complexity. The propulsion subsystem uses a single propellant tank as cg/cp offset control is no longer required. Thermal control is easier with the deployed arrays being self temperature controlled at ~70 degrees C and the spacecraft subsystem and instrument modules each using dedicated louvers for independent temperature control. This design no longer requires the sun-avoidance yaw maneuver.

This conceptual design is patterned after the Option 4 TIMED-H spacecraft which was designed prior to the Option 3.

Option 4

The option introduces the concept of using one small (SMEX or small explorer class) spacecraft and one full-size spacecraft. This option requires the SMEX to fly in a 30 degree inclination circular orbit. The TIMED-H spacecraft is in a 70 degree inclination elliptical orbit and has a minimum perigee altitude of 175 km. The TOMS (a derivation of the off-the-shelf EAGLE spacecraft) spacecraft, with few exceptions, met the mission requirements, so the study team baselined it as the starting spacecraft design for this option. Figure 5-74 depicts the small spacecraft design.

TIMED-H FOR OPTIONS 3 AND 4

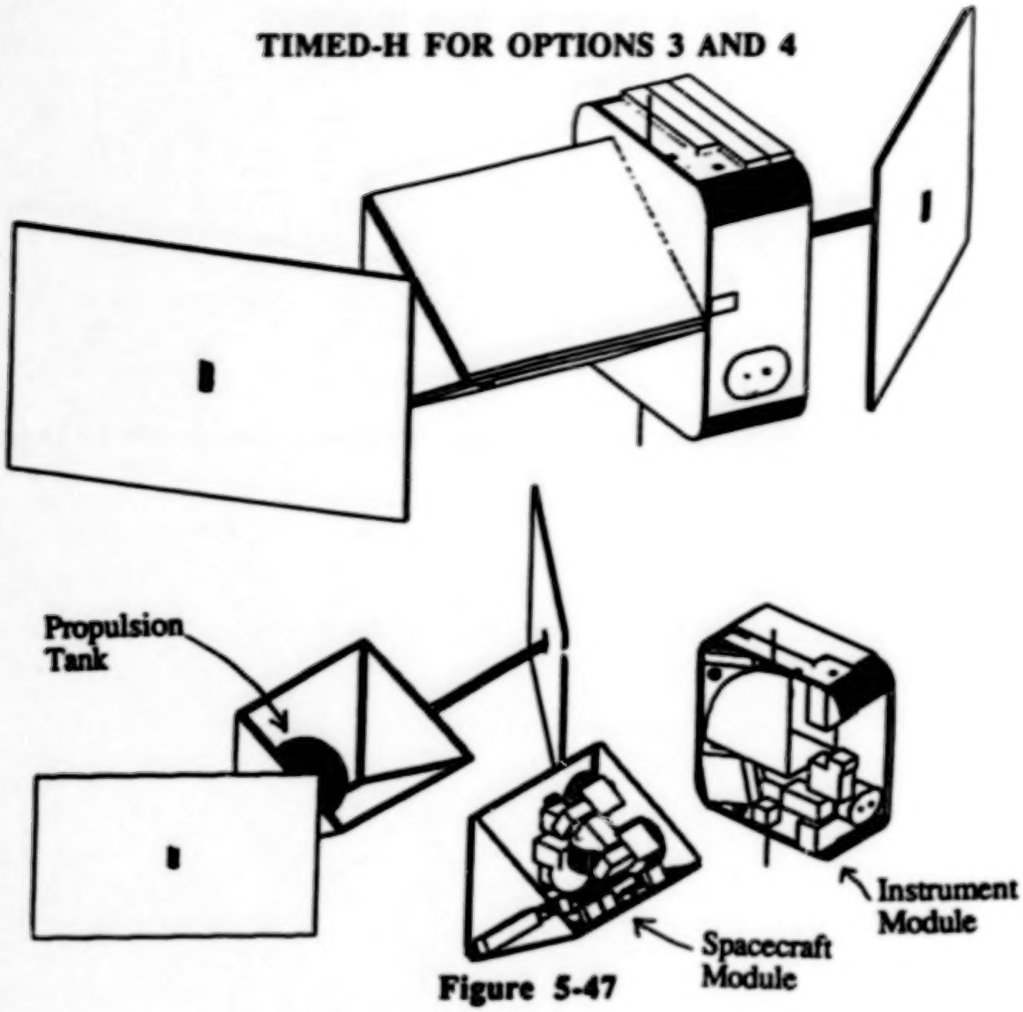


Figure 5-47

TIMED-L FOR OPTION 4 (SHOWING INSTRUMENT VIEW ANGLES)

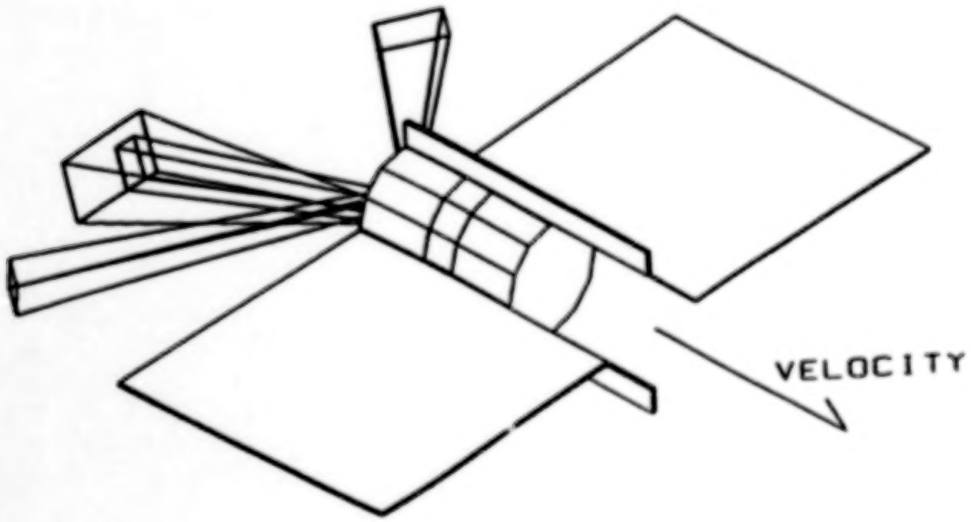


Figure 5-48

Option 5

The TIMED-H spacecraft is developed as described in Option 3 above. The requirements for this one spacecraft are essentially the same as the Option 2 TIMED-H spacecraft. The final design is identical to the Option 2 TIMED-H design except the spacecraft size is 2.0 m in diameter and 2.3 m in length and the propulsion subsystem uses 6 oval 56 cm x 81 cm tanks. Figure 5-49 is a cutaway of the Option 5 single spacecraft design without the outer structure and solar arrays.

OPTION 5 SPACECRAFT CUTAWAY SHOWING ARRANGEMENT OF INSTRUMENTS AND PROPULSION SYSTEM

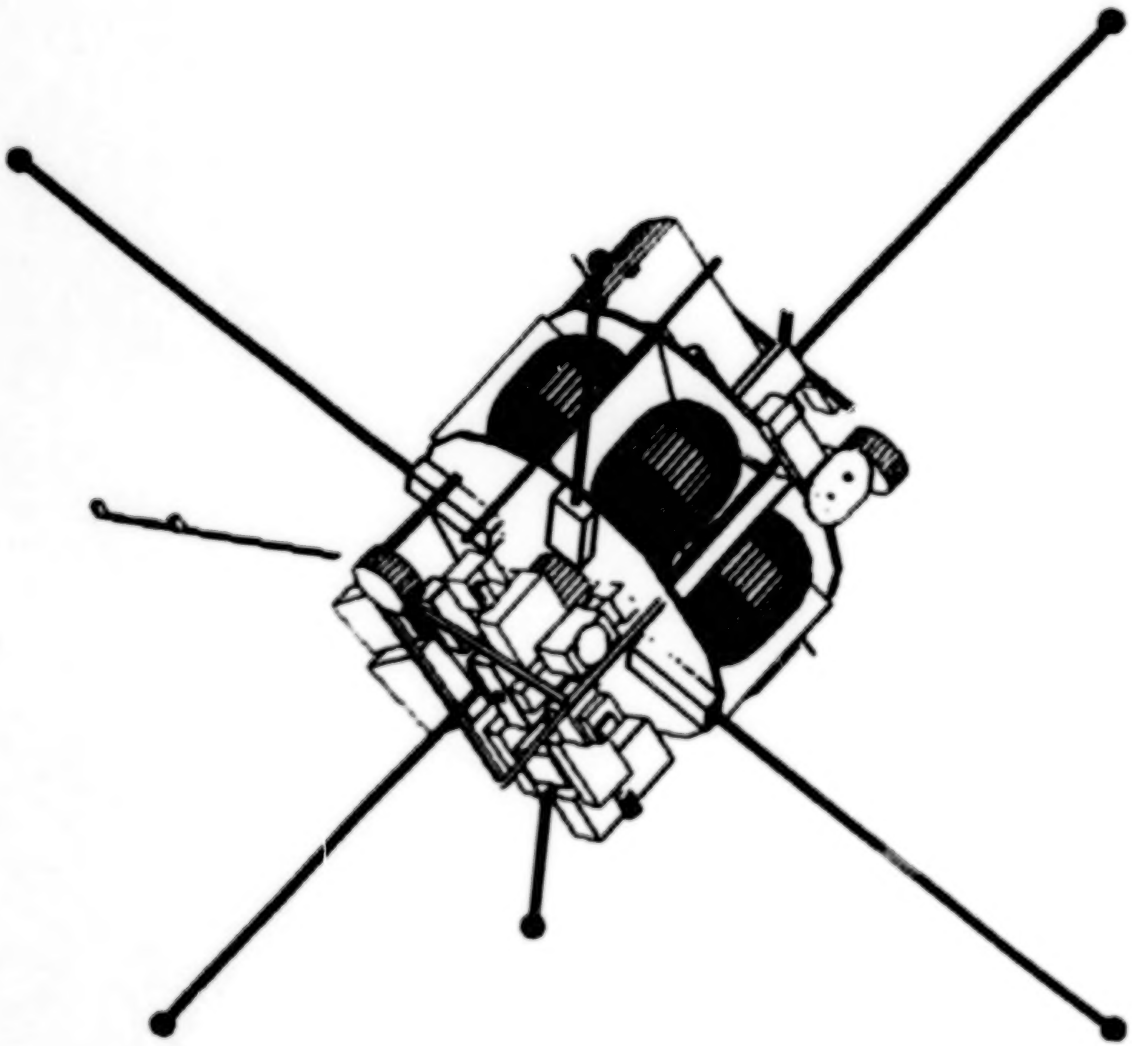


Figure 5-49

6.0 INTEGRATION AND TEST

The integration and test (I&T) plans call for the assembly and test of two observatories in parallel. Because of the identical spacecraft designs, certain economies can be realized in sharing test equipment, fixtures and facilities. The overview of the integration and test flow is shown in Figure 6-1. Scheduling priority is given to the first observatory (shown as TIMED-H in the figure). The second observatory will follow the first by 3 months. If hardware is available for both observatories, it will be installed and tested first on TIMED-H with installation on TIMED-L following as time permits. If necessary, hardware scheduled for TIMED-L may be used in TIMED-H to maintain schedule. The modularity of the design permits some parallel integration of spacecraft and instrument modules in the early stages of I&T.

The integration and test of both TIMED spacecraft starts with the delivery of the subsystem components and instruments to the spacecraft integration facility. Prior to this point, the functional performance of each of the subsystems, instruments or components is tested through their expected environments (i.e., vibration, thermal/vacuum, EMI/EMC, shock, etc.). In addition, external interfaces are tested to reduce the risk of damaging other components or subsystems upon initial mating. Since some subsystems cannot be verified as full subsystems prior to spacecraft integration, their verification must be done at the spacecraft level.

Because most of the TIMED-H and TIMED-L mechanical ground support equipment (MGSE) is needed at the same time during integration, two sets of MGSE are required. Exceptions are the mass mock-ups for those instruments that fly on both spacecraft and the observatory shipping container. These are items which can be shared by both.

Because of conflicts and the fact that one observatory will be undergoing checkout at the launch site at the same time the other observatory is in integration and test, two sets of electrical ground support equipment (EGSE) are required to support the integration, test and launch of the two observatories. It is assumed that the experimenters will develop the EGSE for their instruments. The Project will provide a simple C&DH spacecraft interface simulator to the experimenters for use during instrument development.

6.1 Integration and Test Flow

Figure 6-2 is an expanded diagram that shows the integration and test flow for one of the TIMED observatories. The subsystems are integrated into the lower deck assembly while, in parallel, the instruments are integrated into the top deck assembly.

The subsystems follow a fairly traditional integration, with the harness and the power subsystem installed first, followed by the installation of the C&DH, the attitude control and the communications subsystems in that order. Once these subsystems are in place and checked out on the lower deck, the triad support frame is installed on the lower deck.

The propulsion subsystem is installed next. The propulsion subsystem is modular and will be qualified in parallel with the spacecraft flow and brought to the spacecraft as a unit. It is attached to the triad support frame.

Once the propulsion system is in place, the center-of-gravity (cg) compensators for the attitude control system are installed around the cg, which is located approximately in the center of the propulsion tanks. The internal heat pipes are also installed at this time because they must run down from the top deck to the lower deck radiators through the triad support frame. Other miscellaneous thermal control hardware is also installed at this time.

TIMED OBSERVATORY INTEGRATION AND TEST OVERVIEW

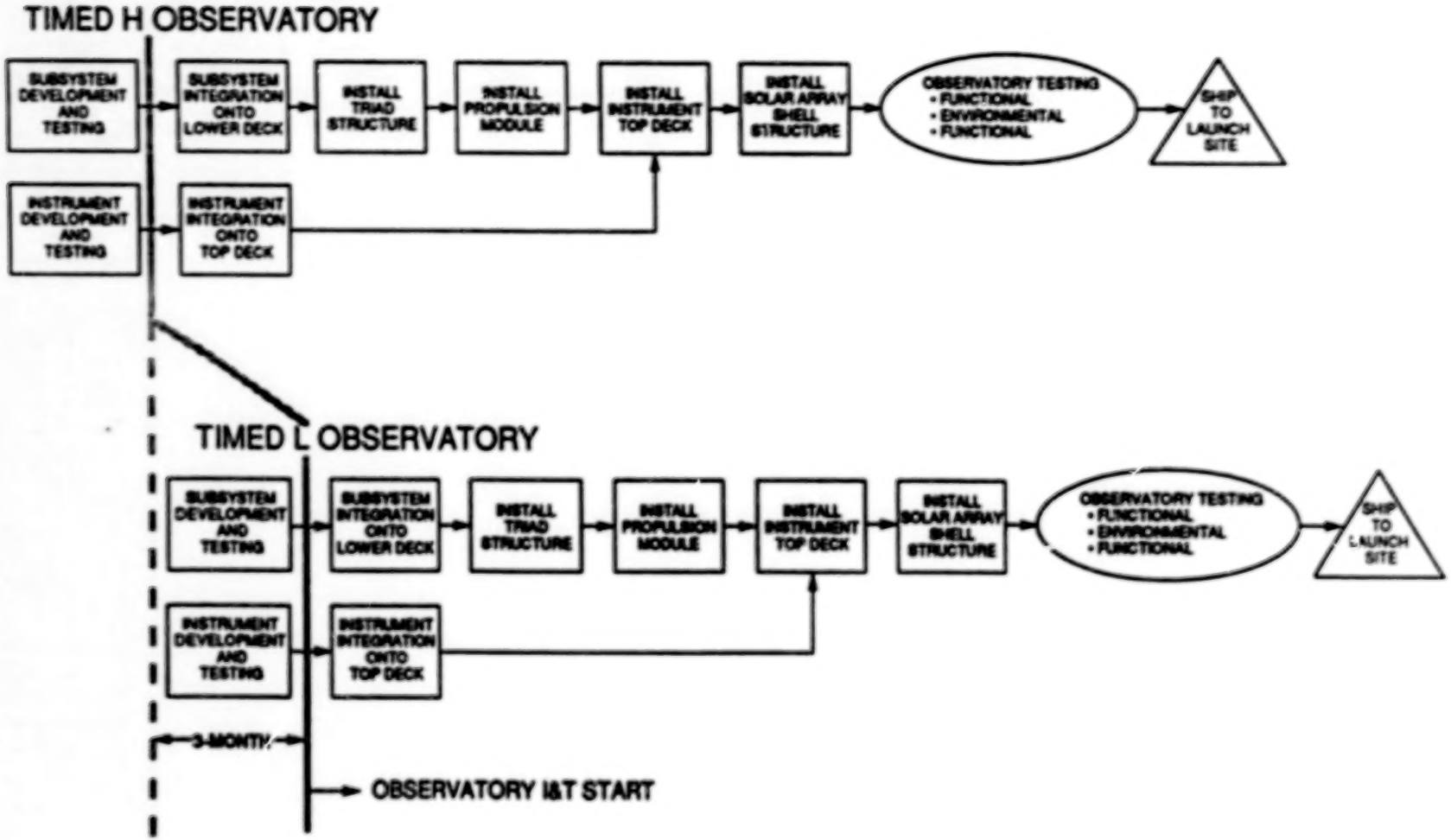


Figure 6-1

TIMED-H OBSERVATORY INTEGRATION AND TEST FLOW

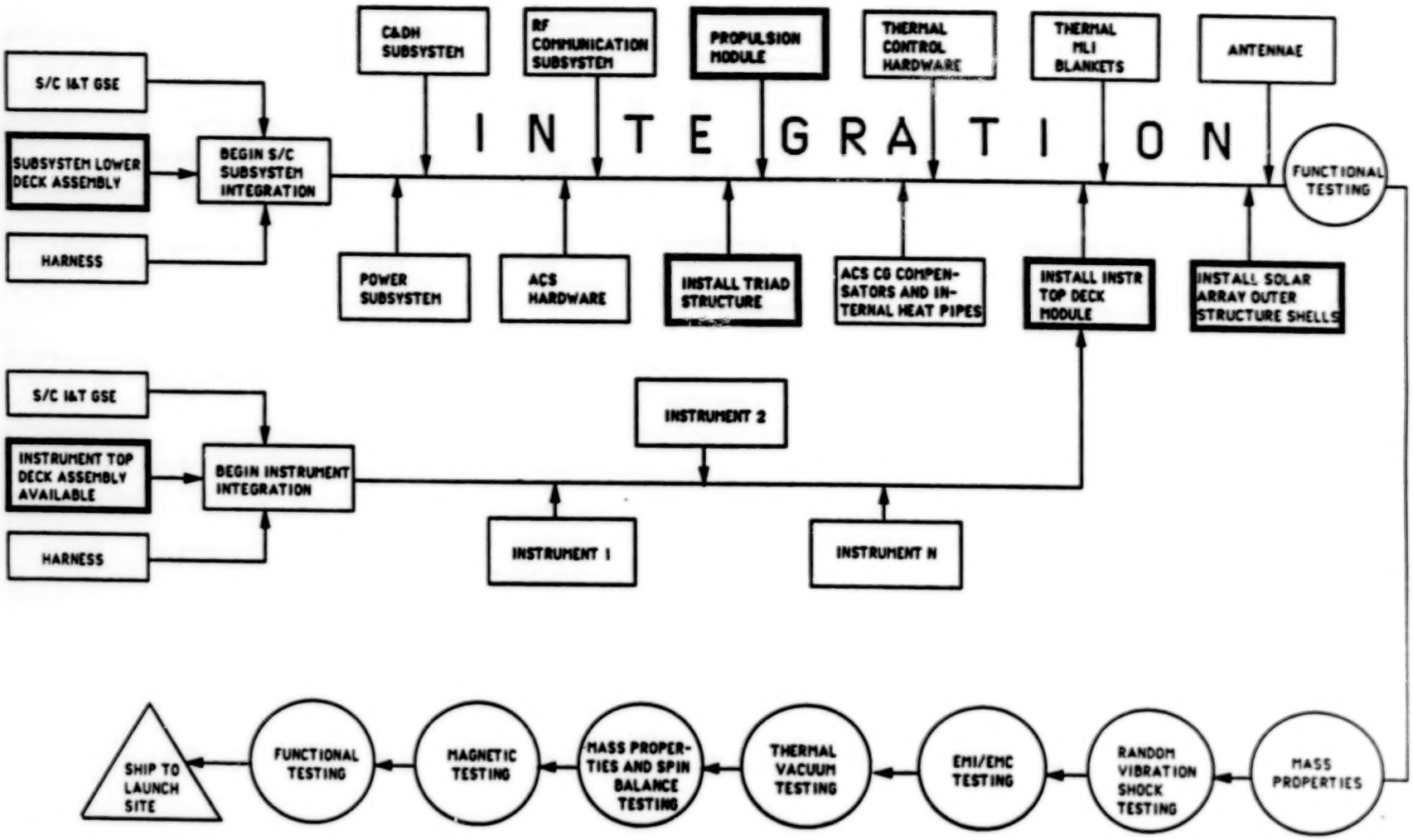


Figure 6-2

At this point, the spacecraft is ready to accept the instrument top deck assembly which had been assembled in parallel with the spacecraft flow. For the first time, the instruments are now connected to the spacecraft power and C&DH subsystems through the flight harness. Prior to this, they had been tested using the power simulator in the EGSE and C&DH spacecraft instrument interface simulators.

The next step is the close-out of the inner structure with multi-layer insulation (MLI) blankets which thermally isolate the internal spacecraft structure from the body-mounted solar array, as described in Section 5.2.

The final steps to the integration are to install the three solar array outer structure shells and then mount the two omni antennas to the exterior surface. If this were TIMED-H, the observatory integration would be complete and it would be ready to start functional and environmental testing at this point. TIMED-L has a unique small deployable solar array. It would be installed and checked out at this time. Some type of deployment test would be run on the array before it is caged for environmental testing.

The remainder of the I&T flow is fairly standard. Everything is now in line with shipping to the launch site. Since mass distribution is so critical, particularly on TIMED-H, a spin balance test was added to the flow. This is followed by a magnetic test to measure the field of the full-up observatory.

6.2 Pre-Integration with the C&DH Subsystem

During the Phase A Study it was recognized that the command and data handling subsystem represents a critical interface for all of the instruments, as well as three spacecraft subsystems: attitude control, power and communications. A concept called the "TIMED Development Facility" (TDF) was proposed in order to reduce the risk of integrating these functions for the first time at the spacecraft level.

The TDF would make use of the C&DH flight equivalent unit (FEU) that resides in the Software Development Facility (SDF) discussed earlier in Section 5.8. The FEU provides for a high fidelity simulation of the distributed computing and data handling environment, enabling a thorough exercise of the flight software in the SDF.

The proposed integration scheme would first test the flight C&DH hardware on the SDF. This validates the use of the SDF as a spacecraft software simulator. Next, the C&DH FEU will be installed into the TDF, which now becomes the platform for incremental validation of the other observatory software interfaces. Each of the successive flight components is connected to this facility prior to being installed on the spacecraft. The software performance is measured and used to validate or modify the TDF software simulation of that hardware and software flight component.

The general flow using the TDF is shown in Figure 6-3. The C&DH subsystem, the attitude control electronics (ACE), the power subsystem electronics, the communications subsystem transponders and the instruments are each brought to the TDF to have their software and electrical interfaces tested before they are physically integrated to the spacecraft structure.

With the TDF, the traditional order of integration is modified so that the C&DH subsystem precedes the power subsystem as the first to be electrically integrated to the spacecraft. Normally, the power subsystem is the first to be integrated so that it can provide power to the other components as they are integrated. However, since the power subsystem

controller only has autonomous control of safety critical functions, full performance evaluation requires significant support from the C&DH subsystem. For this reason, and to reduce risk of software and C&DH subsystem interface problems at spacecraft level integration, each component would be electrically integrated with the TDF. In order to build high confidence in the TDF simulation, the C&DH would be first integrated with its software verifying various software simulations of the C&DH hardware. The C&DH will then provide the command, control and data acquisition function for other subsystems and instruments in the integration process. The C&DH components will have already completed functional and environmental qualification tests before they are brought to the TDF. An integrated subsystem functional test will be run once the C&DH is installed on the TDF. The C&DH will then be integrated to the spacecraft using the spacecraft power simulator and another functional test is run there.

Next, the flight power supply electronics (PSE) is sent through TDF testing along with other power subsystem flight components, except the solar arrays and batteries. Once they have cleared TDF testing, the power subsystem components are then integrated and functionally verified on the spacecraft.

As shown in Figure 6-4, the power subsystem is followed by the attitude control electronics, along with other attitude control system flight components, the communications flight transponders and the instruments.

At the completion of the integration period, the C&DH FEU reverts back to the SDF to support its continuing role of software validation and maintenance.

The advantage of a TDF is that the flight software and flight hardware are brought together early and problems can be resolved before spacecraft integration. Neither the software nor the hardware has to rely on simulators for their pre-integration testing. In fact, the simulators themselves are validated through this approach.

The concept of a TDF will be further explored in Phase B. There are questions of schedule and whether this added step in the integration flow actually adds to the integration time and costs. On the other hand, by combining the functions of the SDF and the TDF perhaps some test equipment and simulators can be eliminated, resulting in cost savings.

INTEGRATE, LAUNCH & CHECKOUT OF TIMED OBSERVATORY

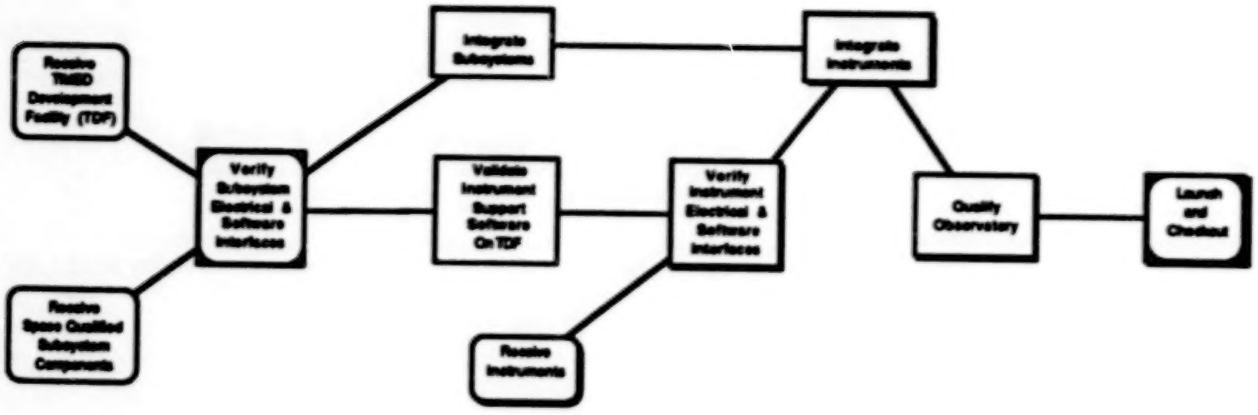


Figure 6-3

TIMED SUBSYSTEM ELECTRICAL AND SOFTWARE I/F VERIFICATION

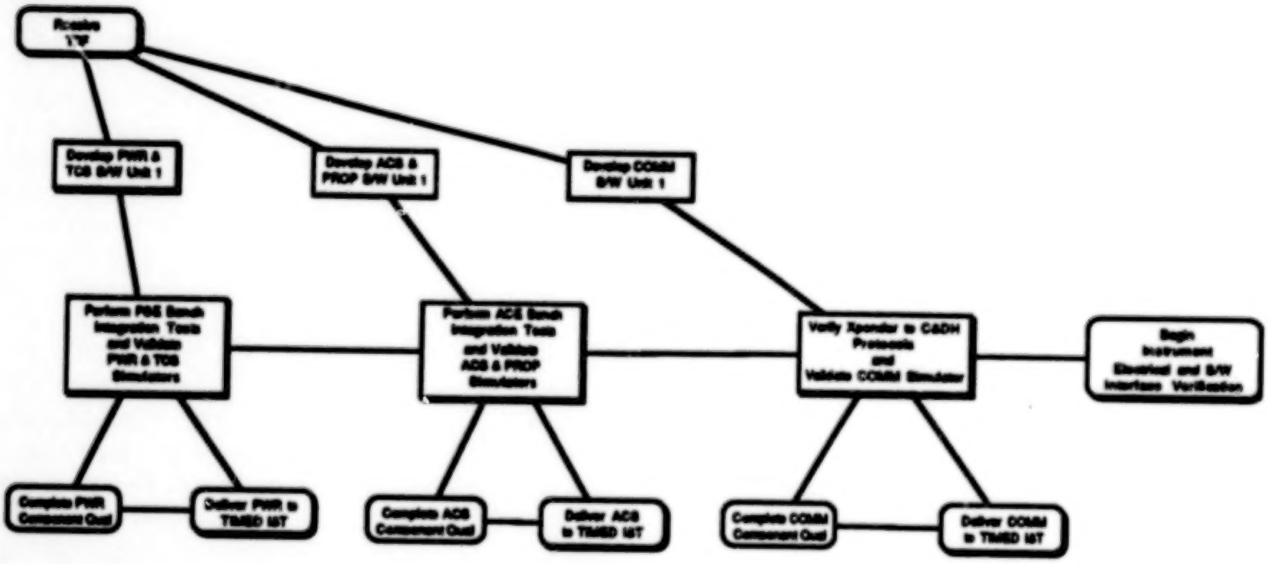

Receive I/F
Develop PWR & TCS S/W Unit 1
Develop ACS & PROP S/W Unit 1
Develop CD&H S/W Unit 1
Perform PDE Bench Integration Tests and Validate PWR & TCS Simulators
Perform ACE Bench Integration Tests and Validate ACS & PROP Simulators
Verify Xponder to C&DH Protocols and Validate CD&H Simulator
Begin Instrument Electrical and S/W Interface Verification
Complete PWR Component Qual
Deliver PWR to TIMED IST
Complete ACS Component Qual
Deliver ACS to TIMED IST
Complete CD&H Component Qual
Deliver CD&H to TIMED IST

Figure 6-4

7.0 SCIENCE DATA SYSTEM

This section addresses the dedicated TIMED ground system that will be used post launch for science operations and science data processing. The GSFC institutional facilities that will be used for the TIMED ground system are described in Section 8.0. Section 7.1 describes the system design considerations that were used to develop this concept. Section 7.2 presents the Phase A strawman ground system architecture. Section 7.3 discusses the GSFC-unique TIMED Ground System. Section 7.4 summarizes the distributed TIMED facilities at the Principal Investigators' (PI's) home institutions.

7.1 System Design Considerations

This section presents an overview of the ground system study methodology that was employed in Phase A and the study results. Details of the ground system study are contained in Reference 5. The ground system study group was part of the overall Phase A Study effort, although it worked independently of the spacecraft subsystem engineering effort. The study approach was to analyze the mission requirements, perform functional analyses, develop functional allocations, and finally perform feasibility analyses to determine the optimal architecture for the ground system.

The study utilized the strawman instrument payload for ground system sizing. The three volumes of the Science Definition Team Report were used as the source of the requirements [Reference 2]. These requirements were augmented based on meetings and discussions with potential PI's and the personal experience of the Ground System Project Team.

Data flow analysis was used as a technique to perform functional analyses. Data flow diagrams and the associated data dictionary helped to form an early understanding of what functions the ground system needs to perform and how these functions interact.

N-squared analysis was used as a tool to help in the definition, tabulation, design, analysis and description of functional and physical interfaces and their relationships. Functional complexity and interface weighing were used in conjunction with the N-squared analysis.

Finally, feasibility analysis was used to evaluate alternative architectures. Alternatives for the feasibility analysis were developed using the outputs of the functional analyses, lessons learned from other programs and programmatic constraints. Advances in computer and mass storage technology were also considered. The twelve selected architectures were evaluated based on their unique characteristics by a set of selection criteria in order to select the best Phase A architecture for the TIMED Science Data System. This architecture is described in the following sections as part of the overall TIMED Ground System.

7.2 Phase A Ground System Overview

Figure 7-1 illustrates the TIMED Ground System. It consists of three parts. First, there is the institutional ground system provided by the NASA Office of Space Communications (Code O), which is unshaded on the diagram. The Code O facilities are described in Section 8.0. Second, there are the TIMED unique facilities at GSFC. Finally, there are the PI facilities located at their home institutions connected to GSFC through existing communications networks, such as the Space Physics Division NASA Science Internet (NSI) Network.

TIMED MISSION OPERATIONS

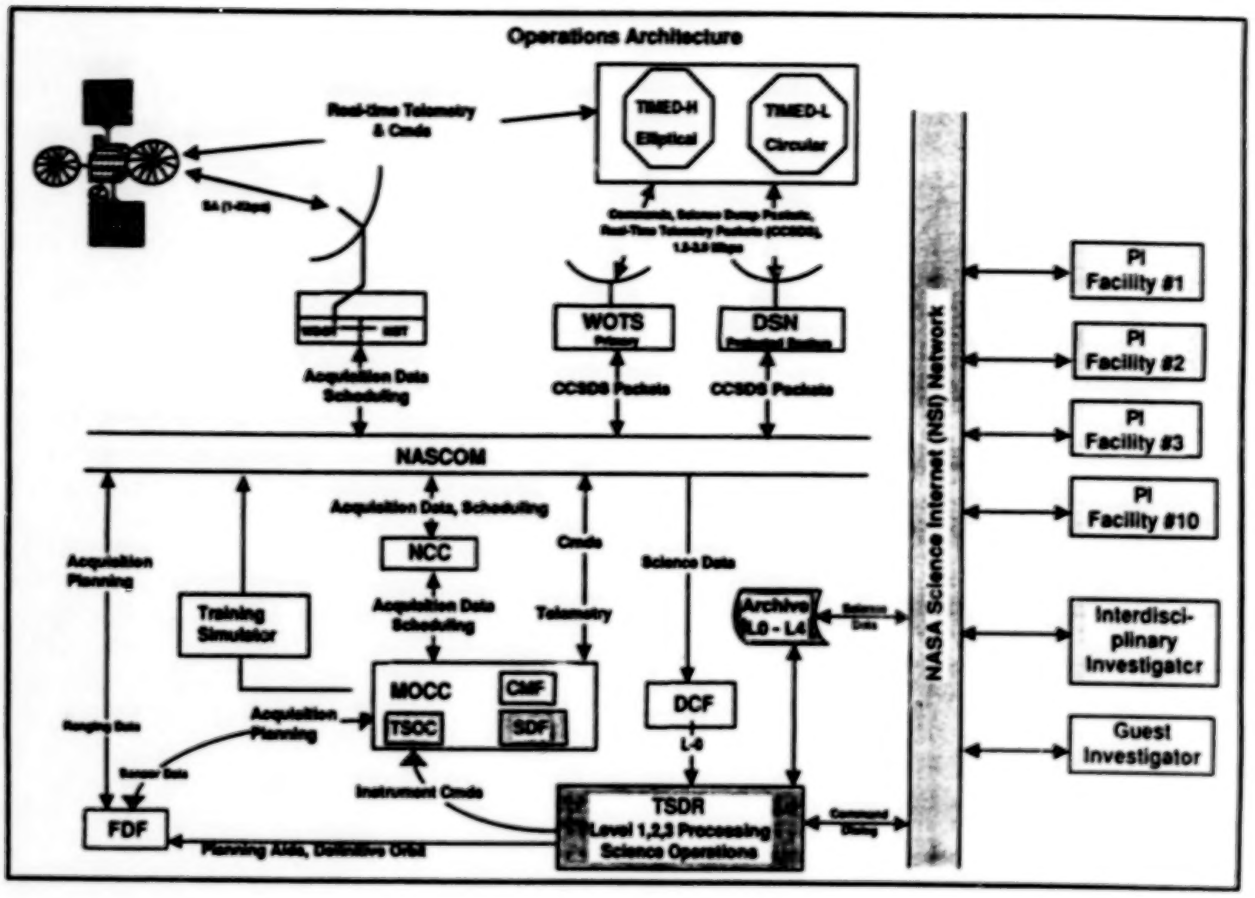


Figure 7-1

7.3 TIMED Science Data Room

The TIMED Science Data Room (TSDR) is the GSFC facility unique to TIMED ground data processing. It serves the functions described below.

7.3.1 Science Planning

The science data system architecture calls for all science planning to be performed in the TSDR. Science planning consists of both long range planning and short term planning. This includes planning TIMED orbits, determining when instruments are duty cycled, correlating ground observations and planning maneuvers, such as the yaw maneuver and orbit adjusts.

7.3.2 Science Operations Scheduling

This activity interfaces with the short term planning described above. The direct result is instrument command requests that are output from the TSDR to the Mission Operations Control Center (MOCC) prior to uplink. Constraint checking is part of this function.

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7.3.3 Instrument Monitoring

Instrument monitoring is another function performed by the TSDR. This consists of monitoring instrument health and safety during a real time TDRSS contact, as well as monitoring instrument performance from science playback data. Trend analysis will also be performed on each instrument.

7.3.4 Science Product Generation

The TSDR is sized to perform all routine data reduction. This consists of Level 1, Level 2 and Level 3 science product generation for all TIMED instruments. The algorithms for the science data processing will be provided by the PI's. The TSDR software will provide standard software environments and standard data access routines.

7.3.5 Science Data Storage and Distribution

The TSDR will provide for the TIMED science archive. The archive is sized at 2 terabytes (TB) for a two year mission using the calculations shown in Table 7-1. This archive sizing assumes that Level 4 model output results are stored in compressed format. The TSDR provides hardware and software to accommodate the science data storage and distribution.

7.3.6 Data and Information Management

The TSDR will also provide the hardware and software needed to perform data and information functions for the TIMED Ground System. This includes services to determine data quality, query, security, catalog, backup and copy.

7.3.7 Candidate TSDR Architecture

A sample architecture for the TSDR is shown in Figure 7-2. The configuration is workstation based with one product generation workstation allocated to each instrument. Table 7-2 contains the sizing calculations for the TSDR.

7.4 PI Facilities

There are two types of Principal Investigators: Instrument Providing PI's and Interdisciplinary PI's.

7.4.1 Instrument PI's

Principal Investigators that supply experiment hardware naturally have a heavier involvement in the day-to-day operations of the ground data system than the interdisciplinary investigators since they are interested in the health and safety of their instrument, as well as the analysis of the data. The instrument PI's will perform some, or all, of the following functions at their facility:

Algorithm Development - The PI is responsible for providing the algorithms that are used to transform the raw instrument counts into final science products gridded in time and space. These algorithms are coded into software at the PI facilities with the help of software standards and data standards provided by the TIMED Project.

Science Planning: Science planning associated with an instrument can be performed at the PI facility. Command requests for instrument observations can originate from the PI facility and then be sent to the TSDR for merger into the overall science plan.

Instrument Monitoring: Although instrument monitoring is performed at the TSDR, it can optionally be performed at the PI facility as well.

Science Product Generation: Science product generation can be optionally performed at the PI facility, provided that the science data is returned to the central archive.

Level 4 Processing: All Level 4 processing for instrument data is performed at the PI's facilities. No computers have been allocated in the TSDR to perform any analysis on instrument data. The Level 4 data will be stored in the TIMED archive.

7.4.2 Interdisciplinary PI's

Interdisciplinary PI's are interested in theory and modeling, using data from more than one TIMED experiment. They have no direct involvement in any instrument operation.

All modeling done by the interdisciplinary PI's will be done at the PI facilities. No computers have been allocated in the TSDR for this function. The model results will be returned to the TIMED archive for storage.

ARCHIVE SIZING

SCIENCE DATA IN KILOBITS PER SECOND		
TIMED-H	19.016	
TIMED-L	10.296	
TOTAL	29.312	
DAILY SCIENCE DATA STORAGE REQUIREMENTS		
Data Type	Expansion	Daily Volume
L0	1.15	.36 GB
L1	1.5	.55 GB
L2	1.1	.60 GB
L3	1.1	.66 GB
L4	1.0*	2.83 GB
TOTAL		2.83 GB
TOTAL ARCHIVE SIZE FOR 2 YEARS IS 2.068 TB		
*Assumes model results are stored in compressed format.		

Table 7-1

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TIMED SCIENCE DATA ROOM CANDIDATE ARCHITECTURE

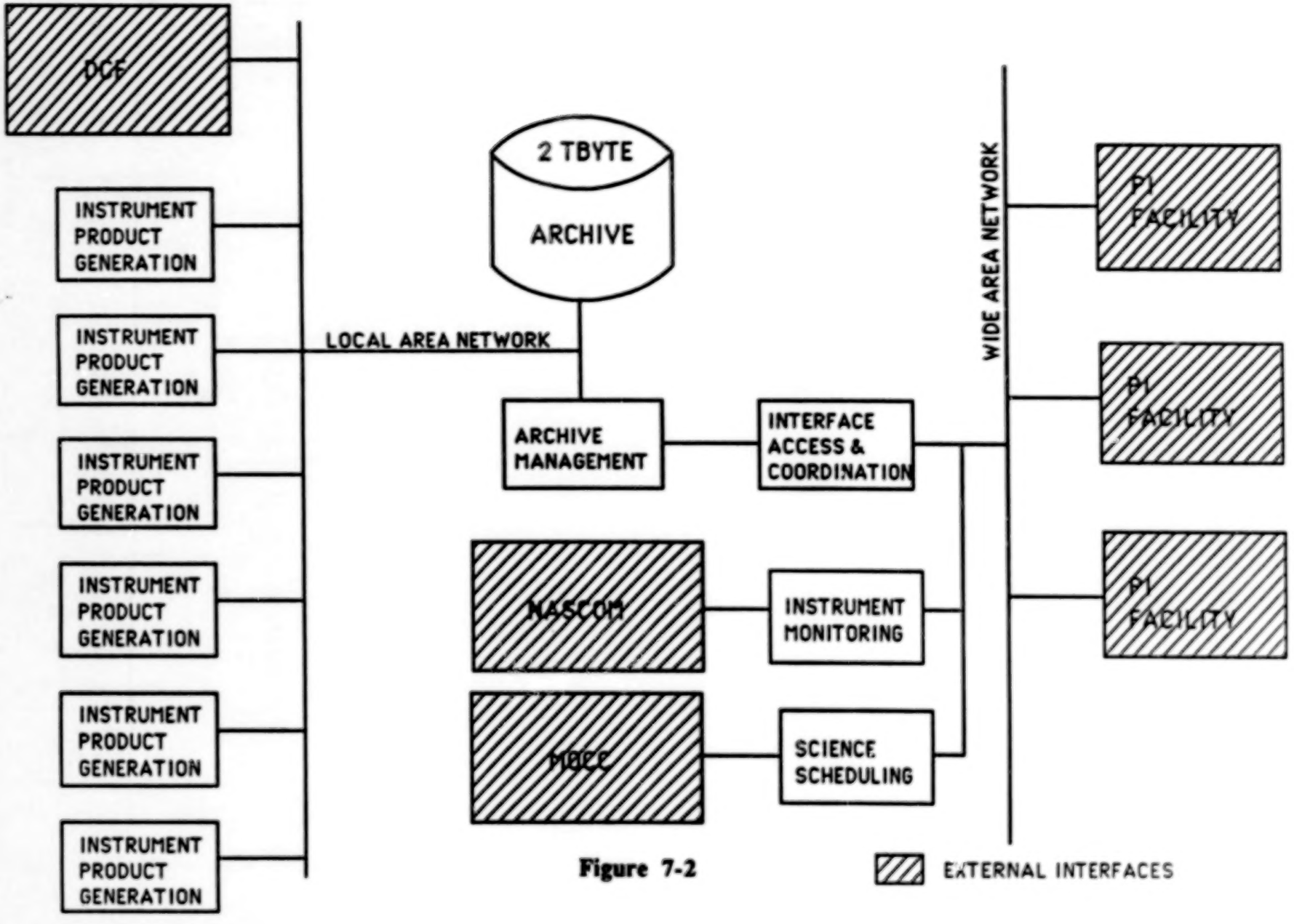

Figure 7-2

TSDR DATA PROCESSING SIZING

		bits	Estimated daily bits	FLOPS per element 12 to 16 bit Conv.	SUM Processing Elm.	LEVEL 0 10 Include Repro	LEVEL 1 30 40 Include Repro	LEVEL 2 60 100 Include Repro	LEVEL 3 90 100 Include Repro
Fabry-Perot Interferometer	1	2,560	221.18E+06	294.91E+06	29.49E+06	153.60E+02	4.61E+04	9.22E+04	13.82E+04
UV Spectrometer	2	64	552.96E+04	737.28E+04	73.73E+04	384.00E+00	1.15E+03	2.30E+03	34.56E+
Imaging Photometer	3	2,560	221.18E+06	294.91E+06	29.49E+06	153.60E+02	4.61E+04	9.22E+04	13.82E+
Near Infrared Spectrometer/Photometer	4	256	221.18E+05	294.91E+05	29.49E+05	153.60E+01	4.61E+03	9.22E+03	13.82E+03
Global UV Airglow Imager	5	11,000	950.00E+06	126.72E+07	12.67E+07	660.00E+02	1.98E+05	3.96E+05	59.40E+04
Neutral Mass Spectrometer/Wind & Temp.	6	3,072	265.42E+06	353.89E+06	35.39E+06	184.32E+02	5.53E+04	1.11E+05	16.59E+04
Ion Mass Spectrometer	*7								
Ion Drift Meter/Retarding Potential Ana.	8	4,400	380.16E+06	506.88E+06	50.69E+06	264.00E+02	7.92E+04	1.58E+05	23.76E+04
Langmuir Probe	9	2,000	172.80E+06	230.40E+06	23.04E+06	120.00E+02	3.60E+04	7.20E+04	10.80E+04
Energetic Particles Analyzer	*10								
Vector Magnetometer	*11								
Solar EUV Spectrometer	12	400	345.60E+05	460.80E+05	46.08E+05	240.00E+01	7.20E+03	1.44E+04	21.60E+03
Satellite Electrostatic Triaxial Accelerom.	*13								
Electric Field Detector/Plasma Wave Exp.	*14								
Infrared Limb Sounder	15	3,000	259.20E+06	345.60E+06	34.56E+06	180.00E+02	5.40E+04	1.08E+05	16.20E+04
TOTAL					337.67E+06	175.87E+03	527.62E+03	105.52E+04	1.58E+06

	ID NO.	CPU Assignment	CPU ID	Level 1-3 CPU Performance
Fabry-Perot Interferometer	1	1	1	29.18E+04
UV Spectrometer	2	2	2	82.08E+03
Imaging Photometer	3	3	3	29.18E+04
Near Infrared Spectrometer/Photometer	4	2		Part of 2
Global UV Airglow Imager	5	4	4	125.40E+4
Neutral Mass Spectrometer/Wind & Temp.	6	5	5	35.02E+04
Ion Mass Spectrometer	7	6	6	
Ion Drift Meter/Retarding Potential Analy.	8	7	7	501.60E+03
Langmuir Probe	9	8	8	22.80E+04
Energetic Particles Analyzer	10	9	9	
Vector Magnetometer	11	10	10	
Solar EUV Spectrometer	12	2		Part of 2
Satellite Electrostatic Triaxial Accelrom.	13	10		Part of 10
Electric Field Detector/Plasma Wave Exp.	14	11	11	
Infrared Limb Sounder	15	12	12	34.20E+04
TOTAL				3.34E+06

Assumptions:
- 10 bits per processing element
- Simultaneous processing, re-processing, and development

* Not part of the selected instrument set.

Table 7-2

8.0 OPERATIONS

This section provides a summary of the Phase A Study preliminary operations concept. The operations objectives are presented first. The architecture as related to operations is then reviewed in terms of four elements: flight, space, ground and science. Finally the operations concepts are presented. The concepts demonstrate how the operations objectives are supported by the proposed architecture.

8.1 Operations Objectives

The goal of operations is to define the spacecraft and instrument operations requirements and to ensure their compatibility with supporting network and ground system capabilities.

8.1.1 Mission Operations Objectives

- Minimize risk to spacecraft health and safety,
- Maximize spacecraft operations through a constant awareness of the state of on-board spacecraft processes,
- Maximize science data acquisition, and
- Maximize support of science activities.

8.1.2 Science Operations Objectives

- Optimize science data acquisition,
- Timely response to Targets-of-Opportunities (TOO's),
- Rapid, efficient processing of acquired instrument data with minimal reprocessing,
- Timely distribution of processed data to investigators' facilities, and
- Safety of processing algorithms and processed data through proven archival practices.

8.2 Operations Architecture

The mission operations architecture (Figure 7-1) consists of the flight element, the space elements and the necessary ground and science elements to support the TIMED Mission. These key elements are shown schematically in Figure 8-1 below.

OPERATIONS ARCHITECTURE ELEMENTS

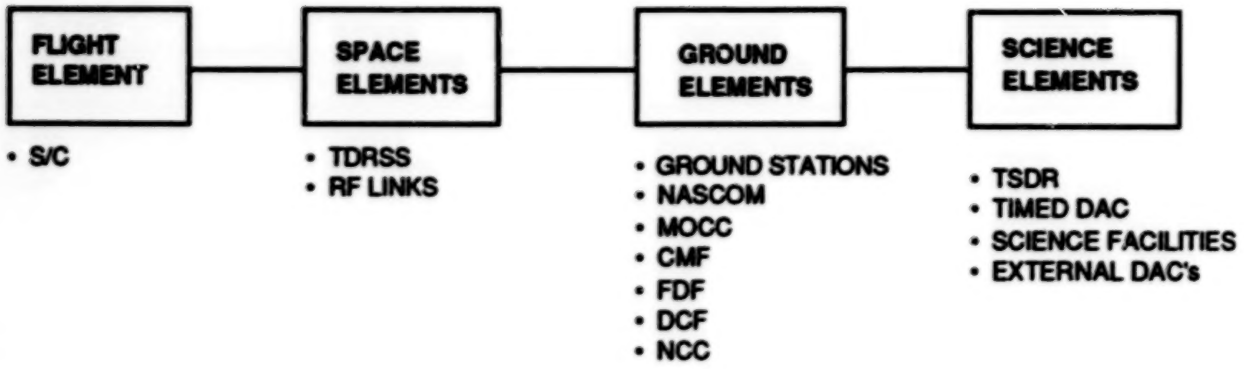

Figure 8-1

8.2.1 Flight Element

The TIMED spacecraft will support the following functions:

- Spacecraft commanding,
- Spacecraft engineering (health and safety) telemetry
- Science telemetry , and
- Tracking and ranging data acquisition.

Communications links between the spacecraft and the ground will be used for commanding and spacecraft telemetry. Science data will be recorded on-board the spacecraft and downlinked to the Wallops Orbital Tracking Station (WOTS). WOTS services will be augmented by other ground stations, as necessary, to ensure the capture of the science data. Candidate data format, data rates, and required acquisition times are discussed in Paragraph 8.2.2.2.

TDRSS and ground tracking services will be used to provide ranging accuracies within the science requirements. These requirements are:

- Orbital position is to be known within 2 km along-track, 1 km radial, and 4 km cross-track.

- Orbital velocity is to be known within 1 m/sec and made available as ancillary data.

- Spacecraft altitude is to be known within 1 km.

- Spacecraft position and velocity knowledge must be derivable, within requirement guidelines, for any orbital position.

8.2.1.1 Spacecraft Communications Subsystem

The TIMED communication subsystem will consist of two transponders: a TDRSS and a Ground Stations Tracking and Data Network (GSTDN) transponder. These transponders allow spacecraft contacts through TDRSS and the Ground Network (GN). (The Ground

Network is used here in its generic sense to include WOTS, Deep Space Network (DSN), and other compatible ground terminals.)

The GSTDN transponder receiver will be on at all times while its transmitter will only be turned on for science data dumps to a ground terminal. The TDRSS transponder will only be turned on for TDRSS contacts. This scenario satisfies the communications link requirements and achieves a lower average power consumption. This scenario poses greater operational risks in that the GSTDN transponder cannot be used as a backup for the TDRSS transponder. The limitation is because the GSTDN transponder can only communicate with ground stations, whereas the TDRSS transponder, which has two communication modes, can "talk" with either TDRSS or a ground station.

8.2.2 Space Elements

The TIMED space elements will consist of TDRSS and ground-to-satellite RF links. This section describes TDRSS use and requirements and factors impacting the TIMED-to-ground RF links.

8.2.2.1 Tracking and Data Relay Satellite System

TDRSS will be used by TIMED for the command uplink, downlinked spacecraft engineering and tracking data, and ranging.

8.2.2.2 Communications Links

Uplink to the TIMED spacecraft will be through TDRSS with the WOTS/DSN as backup. Satellite engineering and tracking data will be downlinked through TDRSS with the WOTS/DSN as an alternative downlink path. Science data will be downlinked through WOTS/DSN. Relevant communications requirements are listed below.

8.2.2.2.1 Requirements

Command Requirements

- TIMED commands will be uplinked to the spacecraft by TDRSS or WOTS as a backup.
- All commands will be uplinked from the MOCC and real-time commanding from the MOCC will not be precluded.

Telemetry Requirements

- TDRS single access S-band is to be used for spacecraft command uplink, health and safety downlink, and for orbit determination.
- Wallops Orbital Tracking Station is to be used for science data downlink.
- The daily dump rate will accommodate the maximum instrument operations as defined by the most intense mission phase and the maximum power generation orbital conditions. Special data requirements, such as burst mode, will be accommodated if possible without adding additional subsystem cost, otherwise this requirement will be met by duty cycling the other instrument operations.

- NASCOM shall be used to transmit science data received at the Wallops Orbital Tracking Station (WOTS) to the TIMED Data Capture Facility (DCF). NASCOM shall be used to transmit spacecraft engineering data from TDRSS to the TIMED MOCC. NASCOM will be used to uplink commands from the MOCC to the spacecraft through TDRSS or WOTS as a backup.

- The instrument data rate requirements (assuming all instruments are turned on) are 19.016 kbps and 10.296 kbps, respectively for the TIMED-H and TIMED-L spacecraft. Data rates given in this section are based on the strawman instrument complement identified elsewhere in this report. The actual orbital average total data rate requirement for the instruments will depend on the instruments actually flown and on the instrument duty cycle.

8.2.2.2.2 Data Format

The TIMED Mission will use the Consultative Committee for Space Data Systems (CCSDS)/Advanced Orbiting Systems (AOS) recommendations for telemetry and commanding systems.

8.2.2.2.3 Data Rates

Commanding data rates will be 2 Kbps or less; the downlinked spacecraft engineering data rate will be 1 Kbps. The downlinked science data rate is 1.5 Mbps or 3 Mbps. The average instrument data rates are 19.016 and 10.296 Kbps, respectively for TIMED-H and TIMED-L. Daily (24 hr.) accumulation is shown in Table 8-1.

TIMED Data Accumulation and Required Dump Time

TIMED S/C	Ave Data Rate (S/C + Instr.) (Kbps) (100 % duty cycle)	Ave Data Rate (With 10% CCSDS Overhead) (Kbps)	Daily Accumulation (Gb)	Total Accumulation (With 25 % Margin) (Gb)	Dump Time Required (in minutes) at 1.5 Mbps/3.0 Mbps	
TIMED-H	20.02	22.02	1.90	2.38	26	13
TIMED-L	11.3	12.43	1.07	1.34	15	7.5

Table 8-1

8.2.2.2.4 Required Downlink Contact Times

For the Phase A Study, it was assumed that TDRSS contacts would be approximately10 minutes per orbit (when scheduling permits). This time will be utilized for commanding the spacecraft and also for receiving the 1-Kbps spacecraft engineering and tracking data.

A comparison of required dump times (Table 8-1) with available contact times at WOTS (Table 8-2) indicates that WOTS occasionally cannot provide support for a 1.5 Mbps downlink for all orbital scenarios with instruments operating at 100 percent duty cycle. Contact time with the TIMED-H spacecraft in a circular orbit is insufficient to completely transmit recorded data in this scenario. However, this is not a problem at the 3.0 Mbps downlink or for reduced instrument duty cycles.

WOTS Worst Case Minimum Daily Contact Time

TIMED S/C	Orbit (Km)	Daily Contact Time (min)
TIMED-H	6000 x 150	30
TIMED-H	400 Circular	15
TIMED-L	400 Circular	37

Table 8-2

8.2.3 Ground Elements

8.2.3.1 Ground Stations

Available networks and ground support stations including commercial resources were studied during Phase A. Within the NASA environment, TDRSS, WOTS, DSN, and the Transportable Orbital Tracking System (TOTS) were identified as possible sources of both primary and backup support.

8.2.3.1.1 Wallops Orbital Tracking Station (WOTS)

WOTS will be used to receive the downlinked TIMED science data and as an alternative communication path to TDRSS. The GSFC-WOTS communications link is rated at 1.55 Mbps. All data will be carried over this link. Scheduling of WOTS resources is coordinated through the Spaceflight Tracking and Data Network (STDN) Network Control Center (NCC) at the Goddard Space Flight Center (GSFC).

8.2.3.1.2 Deep Space Network

The DSN is designated as backup to the WOTS for TIMED support. Communication links, however, between the DSN Stations/JPL and GSFC use lower transmissions (224 Kbps) than the spacecraft-to-ground link, so data will not be received at GSFC in real or near-real time. Scheduling of DSN resources is coordinated through the STDN NCC at GSFC.

8.2.3.1.3 Other Support Capabilities

NASA's Transportable Orbital Tracking System (TOTS) as well as the European Space Operations Centre Stations, and commercially managed ground stations were investigated during the Phase A Study. These capabilities should be investigated further if the current NASA (WOTS/DSN) facilities are found to be inadequate for TIMED support.

8.2.3.2 Communications Support Functions

NASA Communications (NASCOM) will provide the transport of TIMED operational data between GSFC and other locations such as WOTS, DSN or the White Sands Complex (WSC). NASCOM services will also include the Voice Swiching System (VSS), CCTV and TELECON.

8.2.3.3 Mission Operations Control Center Functions

The primary MOCC functions are:

- Coordination between mission operations and science operations,
- Mission planning and scheduling,
- Command and control (command validation, constraint checking, and load generation),
- Spacecraft and instrument health and safety monitoring (including telemetry processing and contingency response), and
- Science data recovery.

Other functions include engineering data trend analysis, project database administration, data accounting to assist planning recorder dump operations and off-line processing.

8.2.3.4 Command Management Functions

Command management functions will be provided by the Command Management Services (CMS) which may reside in the MOCC or the Command Management Facility (CMF). CMS will provide command memory management, constraint checking, and command sequence generation. It will aid in development of operation plans.

The CMF can also provide simulation capabilities which include:

- Modelling on-board sensors,
- Modelling the spacecraft environment,
- Generation of simulated engineering telemetry, and
- Command input.

8.2.3.5 Flight Dynamics Functions

The Flight Dynamics Facility (FDF) will provide orbit analysis and support for onboard attitude control. The support will include determination of attitude sensor misalignment and calibration, verification and evaluation of the ACS performance, and attitude maneuvers. Orbit-related data (including maneuver and propulsion data) will be provided to both the MOCC and TSDR; orbital adjustment procedures will be provided to the MOCC.

8.2.3.6 Data Capture Functions

The Data Capture Facility (DCF) will receive the recorded science data and perform Level-0 processing (LZP). LZP includes time-ordering, removal of redundant data, and data reversal (if necessary). The data will then be forwarded to the MOCC, TSDR, and TIMED Data Archive Center (DAC).

8.2.3.7 Network Scheduling and Coordination Functions

The Network Control Center (NCC) will provide network scheduling services to the MOCC. These services will include scheduling and coordination of TDRSS, ground stations and network services required for support.

8.2.4 Science Elements

Science functions have already been addressed in Section 7. However, this section discusses the operations aspects of this important element.

8.2.4.1 Science Operations and Data Processing Functions

Science operations and data processing functions include instrument operations, planning and scheduling, monitoring and science data processing, distribution and archiving.

The TIMED Science Data Room (TSDR) will be responsible for instrument operations, scheduling, and monitoring. The TSDR will also process science data. The TIMED Science Operations Coordinator (TSOC) will serve as the interface to the MOCC for the TSDR and the scientists.

The investigators in the TSDR will build command sequences for their respective instruments. The instrument command sequence list will be forwarded to the MOCC for integration with the spacecraft load prior to uplink.

Instrument quick look (Q/L) data will be available for instrument evaluation. The TSDR will remain cognizant of the spacecraft condition to aid analysis.

Upper level science data products (Levels 1, 2, 3) will be generated in the TSDR. The Investigators will produce the Level-4 products. Data products will be archived at a TIMED DAC. Data processing is discussed further in Paragraphs 8.3.2.4 and 8.3.3.4.

The TSDR will provide to the Principal Investigators (PI's) ancillary data to support science investigations.

8.2.4.2 TIMED Data Archive Center Functions

The TIMED Data Archive Center will receive Level-0 and higher TIMED data products for archiving. Storage of ancillary data will also be provided by this function.

8.2.4.3 Science Facilities/Principal Investigator Functions

Using FDF-provided orbit data, the PI's will plan science data acquisition, determine required commands and ascertain necessary calibrations. The PI's will provide command request lists to the TSDR.

The PI's will interface with the TSOC and the TSDR personnel to resolve instrument and administrative matters. Contact with the MOCC will be via the TSOC.

8.2.4.4 External Data Archive Centers

Data archive centers (DAC's), containing other scientific data, may request TIMED data for their use. At the same time, TIMED Instrument Investigators may request correlative data from external DAC's.

8.2.5 Architectural Element Functional Flow

Figure 7-1 identified the architecture elements relevant to the TIMED Mission and Science operations. Figure 8-2 illustrates the data flow between these elements.

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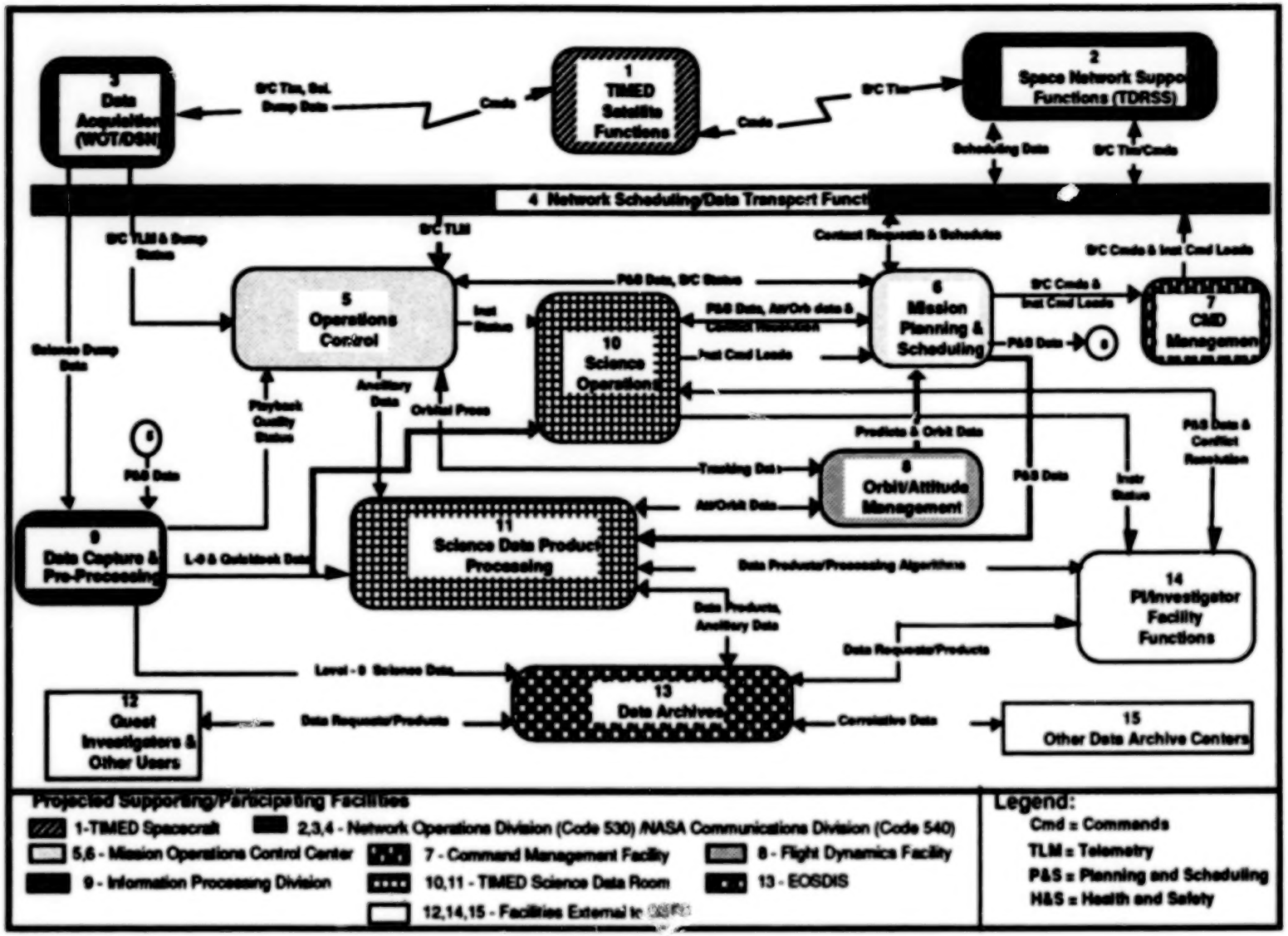


Figure 8-2 TIMED Architectural Element Functional Flow

8.3 TIMED Operations Concept Summary

The TIMED Mission and science operations concepts are presented in this section. The section describes how the architecture and the concepts support the mission operations objectives stated in Section 8.1.

8.3.1 Concept Overview

Operations for both TIMED satellites will be conducted at GSFC in the MOCC and TSDR. The MOCC will serve as the center point for overall satellite control and monitoring, whereas the TSDR will conduct the science operations.

8.3.2 Mission Operations Concepts

The primary Mission operations functions will be mission planning, command and control, spacecraft and instrument health and safety monitoring and science data recovery. Mission operations will also coordinate operations with the remotely located science community through the activities of the TSOC who represents the Investigator's Working Group (IWG).

8.3.2.1 Mission Planning

The MOCC will be responsible for mission planning and scheduling. The function entails integration of science operations plans with mission operations activities, as illustrated in Figure 8-3.

A Long Term Operations Plan (LTOP) will be developed which encompasses the entire mission life cycle for both satellites. The LTOP will identify general mission activities and associated resource requirements. A comparable document, the Long Term Science Plan (LTSP) (Section 8.3.3.1) will be developed by the IWG. The LTSP will describe and prioritize planned science activities). The LTOP and LTSP will be used to develop short term operations plans.

A 28-day planning cycle (Figure 8-4) will be used to develop the short term operations plans. This period coincides with predict coverage from FDF. Briefly, during the cycle the following will occur:

- Available resource levels will be determined by the Flight System Engineers within the MOCC.
- The total resource level available for science activities during the operational period will be determined by the Project Operations Director (Post-launch equivalent of Mission Operations Manager) within the MOCC,
- Resources for science operations will be allocated by the TIMED Science Operations Coordinator (TSOC) in accordance with the LTSP and communicated to the TSDR,
- A Mission Operations Plan (MOP) describing mission activities during the next 28 days will be constructed,
- A Short Term Instrument Plan describing science activities will be constructed in the TSDR and then forwarded to the MOCC,

TIMED PLANNING AND SCHEDULING PROCESS

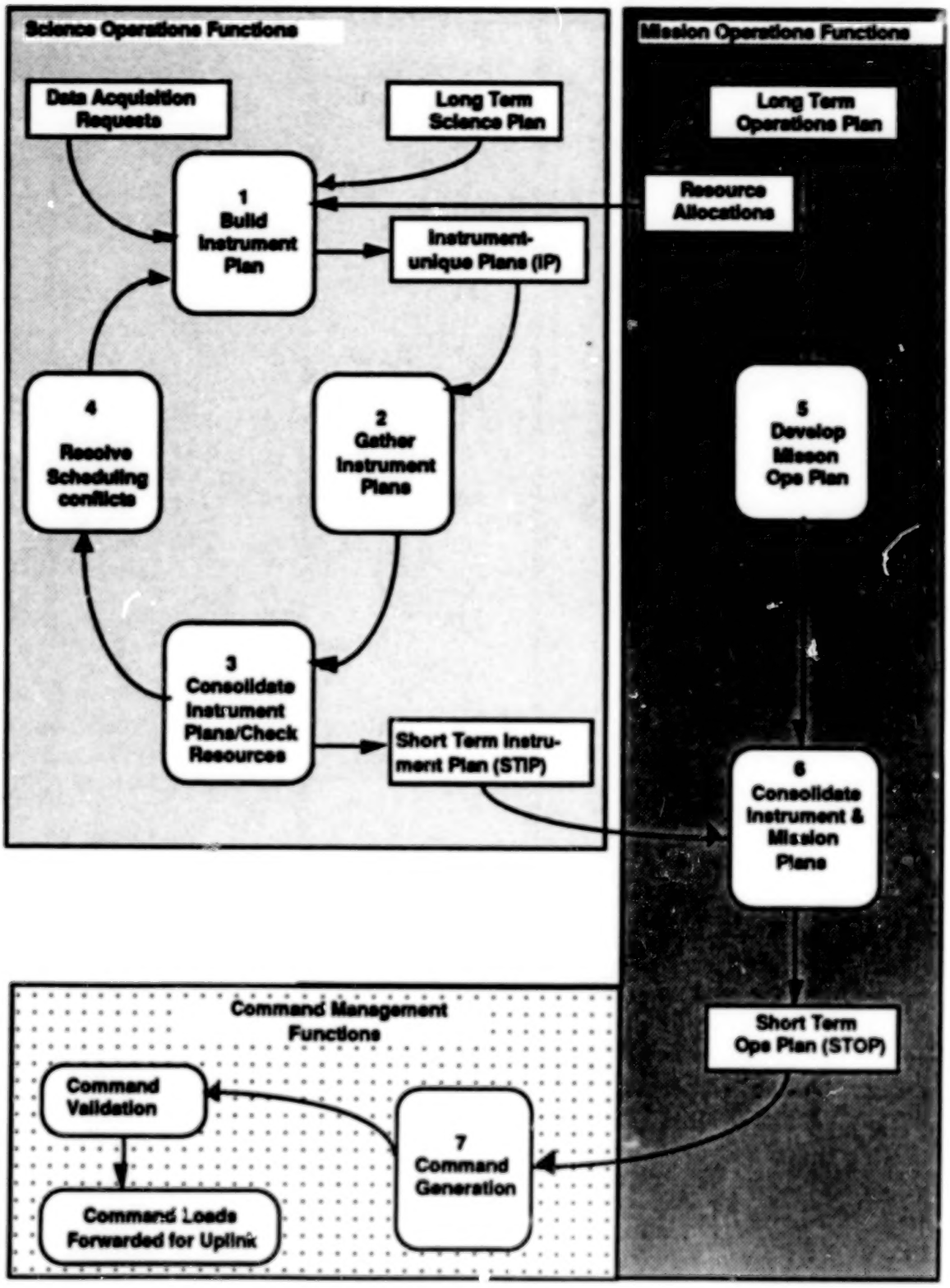


Figure 8-3

- The Short Term Operations Plan (STOP) will be created by integrating the Short Term Instrument Plan (STIP) from the TSDR and the MOP,

- Conflicting instrument operations requests among investigators will be resolved by the TSOC, and

- After conflicts are resolved, the final STOP will be converted by the command management services into a command load.

The STOP will be continually refined to reflect changes in orbital predictions, spacecraft tasks, or science plans such as TOO's. Integration of the MOP and STIP is a joint effort between the MOCC and the TSDR.

SNAPSHOT OF THE PLANNING AND SCHEDULING CYCLE OVER A 28-DAY OPERATIONAL PERIOD

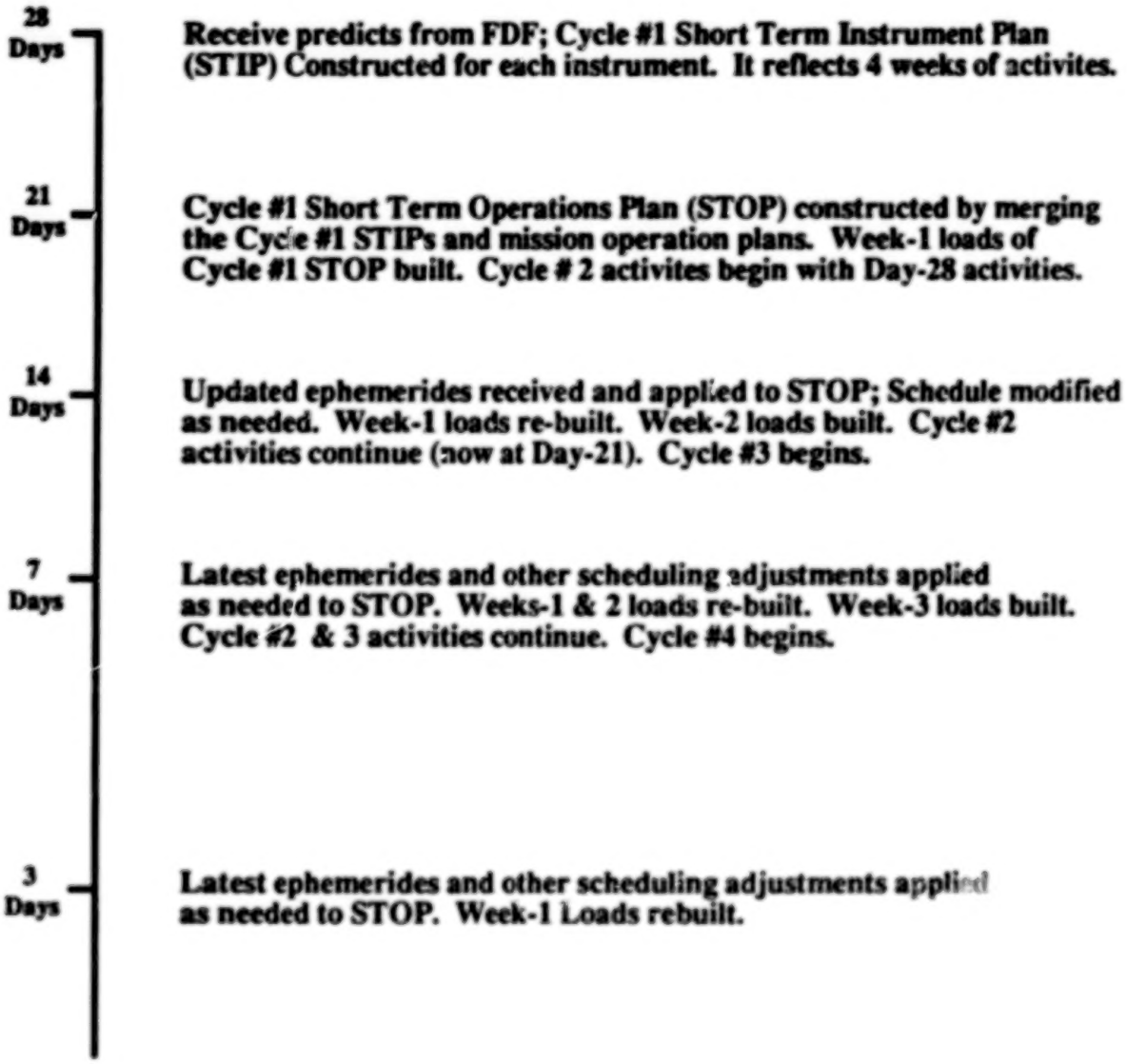

28 Days
Receive predicts from FDF; Cycle #1 Short Term Instrument Plan (STIP) Constructed for each instrument. It reflects 4 weeks of activites.
21 Days
Cycle #1 Short Term Operations Plan (STOP) constructed by merging the Cycle #1 STIPs and mission operation plans. Week-1 loads of Cycle #1 STOP built. Cycle # 2 activites begin with Day-28 activities.
14 Days
Updated ephemerides received and applied to STOP; Schedule modified as needed. Week-1 loads re-built. Week-2 loads built. Cycle #2 activities continue (now at Day-21). Cycle #3 begins.
7 Days
Latest ephemerides and other scheduling adjustments applied as needed to STOP. Weeks-1 & 2 loads re-built. Week-3 loads built. Cycle #2 & 3 activities continue. Cycle #4 begins.
3 Days
Latest ephemerides and other scheduling adjustments applied as needed to STOP. Week-1 Loads rebuilt.

Figure 8-4

8.3.2.2 Command and Control

The MOCC has overall responsibility for satellite commanding. MOCC personnel will build command loads to cover 24 hours of operation using the Command Management Services. Pre-stored commands required by science operations will be incorporated; however, mission operations will be able to deny an uplink request if spacecraft conditions are not conducive to command execution.

Real-time commanding will be used on a contingency basis by the MOCC.

Command loads will be forwarded by the MOCC to the WSC/NGT via NASCOM for uplink. The Space Network will uplink the loads to the TIMED satellites. WOTS/DSN will serve as alternative commanding paths.

8.3.2.3 Monitoring

The spacecraft and instrument health and safety monitoring and spacecraft performance trending will be performed in the MOCC on a 24-hour basis. Instrument performance monitoring will be conducted in the TSDR. Attitude control system telemetry will be forwarded to FDF for tracking and orbital analysis. Responses to anomalous situations will be in accordance with pre-defined operational procedures.

8.3.2.4 Science Data Recovery/Processing

On-board recorders will be played back to WOTS or to alternate ground stations as required. Playback data (and any interleaved real-time data) will then be relayed to the DCF. Recorded data will be viewable off-line only.

The DCF will process science telemetry data to Level-0 and distribute it within 24 hours of receipt to the TSDR and MOCC. The Level-0 data will also be forwarded to a data archive center. Quick look (Q/L) data (which is a subset of the on-board recorded data stream) will be available to the MOCC and TSDR within 2 hours of receipt of the downlinked data.

8.3.2.5 Mission Operations Personnel

Mission Operations functions will be performed by the Flight Operations Team (FOT) which resides in the MOCC. It will operate within guidelines and procedures which are agreed upon by project engineers, scientists and the Mission Operations and Data Systems Directorate prior to launch.

The MOCC will be staffed by the FOT 24 hours per day, 7 days per week. The team's functions (Figure 8-5) are essentially divided into three categories, engineering analysis (Lead Observatory Engineer, Lead Spacecraft Analyst, and the Mission Planner), performance monitoring (Realtime Operations Supervisor and Spacecraft Analysts) and management (FOT Manager, Admin Coordinator). All positions (except the Real-time Spacecraft Analyst slots) are day shift functions. Additional spacecraft subsystem support will be required due to frequent maneuvers and attitude adjustments. This support will be provided by the Engineering Directorate and FDF.

Proposed Flight Operations Team Organization

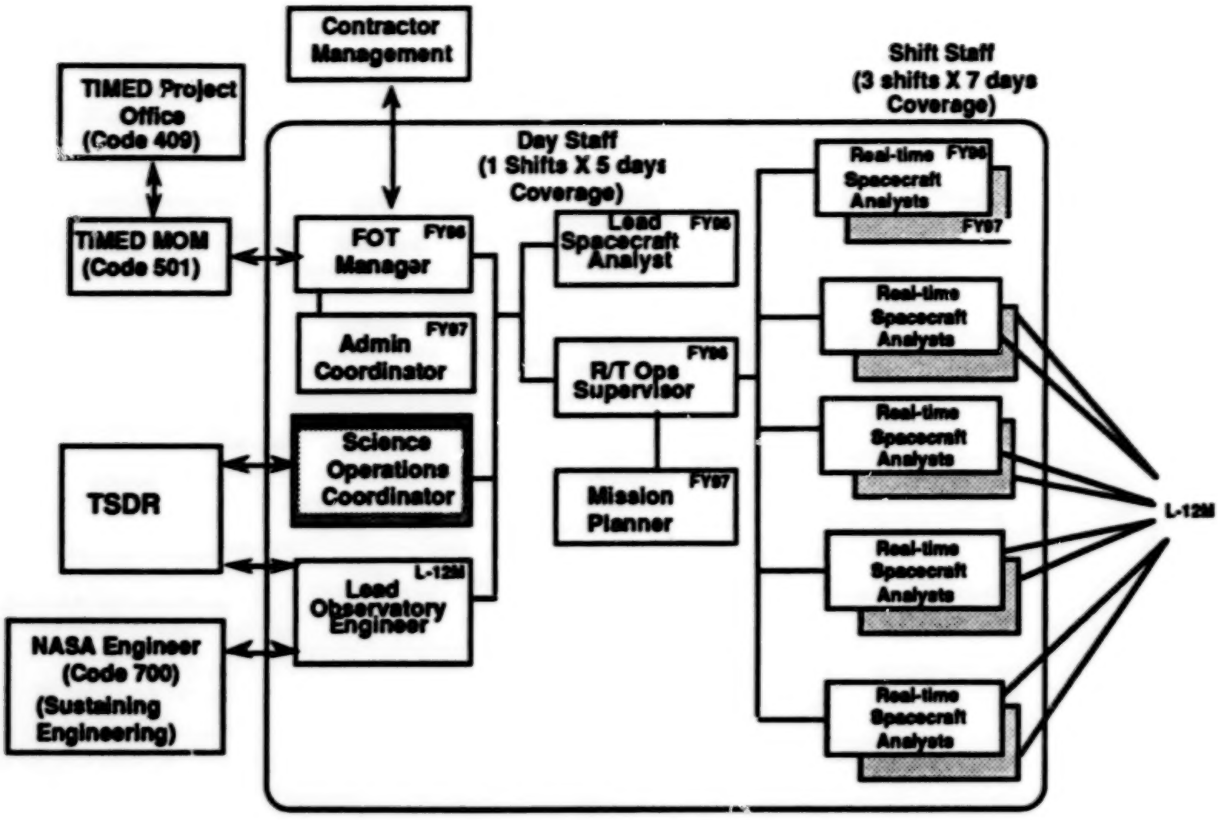


Figure 8-5

The FOT will be phased-in gradually to ensure its effective utilization. The phase-in is structured to involve necessary personnel at critical stages during project development and implementation. Shift analysts will be introduced at one year before launch to use simulations and testing activities to facilitate training.

8.3.3 Science Operations

TIMED science operations for both satellites will be conducted at GSFC primarily within the TSDR and TIMED DAC. Science operations functions will include:

- Instrument planning and scheduling,
- Instrument command and control,
- Instrument monitoring,
- Science data product processing,
- Information access and management,
- Data archiving, and
- Data distribution.

Some planning and scheduling and data product processing activities will occur at the Principle Investigators' facility.

8.3.3.1 Science Planning and Scheduling

PI's will construct operation plans for their instruments. The individual plans will be consolidated by the TSDR into a single plan. The single plan will be forwarded to the MOCC for integration with spacecraft plans. This process will be described in greater detail in subsequent paragraphs and is illustrated in Figures 8-3 and 8-4 in Section 8.3.2.1. Figure 8-3 describes the construction process. The construction timeline is shown in Figure 8-4 .

Spacecraft resources are expected to be limited. Science acquisition/activity priorities will be established by the IWG and integrated into a Long Term Science Plan (LTSP). The LTSP will encompass the entire mission life cycle for both satellites, but will be updated as needed. The LTSP will include mission objectives and priorities to accomplish science objectives.

The TSOC will allocate the resources available for science operations in accordance with the scientific priorities established in the LTSP. Some resource margin or reserve will be set aside for TOO's so as to minimize scheduling disruption.

The planning and scheduling personnel will be advised of their instrument allocations by the TSOC. The PI's will request through the TSDR the desired instrument activities based upon the allocated resource levels. They will construct an Instrument Plan (IP) that covers the associated operational period. The IP will comprise desired science data collections, required instrument maintenance (such as calibrations) and the science plan which describes instrument operations. The IP will be continually refined to reflect changes in orbital predictions, spacecraft tasks, or science plans. Integration of the IPs into a STIP will be performed by the TSDR prior to submission to the MOCC. The TSDR will ensure that there are no conflicting resource requests as well as compliance with resource allocations for science operations. Integration of the STIP with the spacecraft operations plan to form the STOP is a joint effort between the MOCC and the TSDR. Conflicting requests will be resolved by the TSOC; the STIP will be updated accordingly. Requests for data acquisition desired by scientists such as interdisciplinary investigators will be submitted to the appropriate investigator via the TSDR. Such requests will be tracked by the TSDR.

8.3.3.2 Instrument Command and Control

Instrument command sequences will be built using the STIP inputs into the STOP. The instrument command loads will be constructed by Command Management Services. Commands will then be validated to ensure exclusion of undesirable sequences such as instrument checkout commands that are mistakenly issued during the operations phase or syntactically incorrect commands. The instrument command sequence list will be forwarded to the MOCC for integration with the spacecraft command loads.

Instrumenters will also provide to the TSDR pre-stored command sequences which will reside in the MOCC. Such sequences might include safe-hold or mode-switching command sequences. The PI, via the TSOC, can request that the pre-stored commands be uplinked. Uplink of the commands would be contingent upon available spacecraft resources and current status of both the affected satellite and instrument. Use of pre-stored commands will be primarily during contingency operations or for frequently performed activities.

8.3.3.3 Instrument Health and Safety Monitoring

The instrument health and safety monitoring will be performed in the MOCC on a 24-hour basis. Instrument performance monitoring will be conducted in the TSDR during day shift. Instrument quick look data will be available for instrument evaluation. Playback data will be used to assist instrument performance trending by TSDR personnel. The TSDR will remain cognizant of spacecraft conditions to aid analysis.

The TSDR will develop operational procedures to be used in the event of unanticipated operational scenarios such as anomalies or Targets-of-Opportunity so that responses will be in accordance with predefined procedures.

8.3.3.4 Science Data Product Processing

The DCF will process the science and telemetry data to Level-0. Processed data will be distributed to the TSDR within 24 hours of receipt for further reduction. The Level-0 data will also be forwarded to the TIMED Data Archive Center.

The TSDR will generate the Level-1, 2 and 3 data products. (PI's can perform the Level-1, 2, and 3 processing if agreed upon by the TSDR.) Level-4 products and higher will be produced by the investigators and returned to the TSDR for data archiving.

The TSDR will also validate the data products. Product validation will determine if generated data are within expected ranges. The data validation process will be performed by the PI and his staff for those products associated with his instrument.

The TSDR will also provide ancillary data to the PI's to support science investigations.

8.3.3.5 Information Access and Management

The TSDR will provide both information access and management. Data accountability will be necessary to ensure that all desired data was collected and distributed to the appropriate destinations. The TSDR will confirm that data acquisitions were made. It will catalog and then store processed science data. The TSDR will generate metadata and browse data to facilitate searches by the user community (which includes interdisciplinary scientists, students, and guest investigators). Procedures for requesting cataloged data (data distribution requests [DDR's]) will be provided by the TSDR. Requested data will be tracked to ensure that the requests are fulfilled.

The TSDR will also provide procedures for submitting data acquisition requests (DAR's) by guest investigators.

8.3.3.6 Data Distribution

All requests for data (whether they are DAR's or DDR's) will be cataloged and tracked by the TSDR. The DAR's will be forwarded to the appropriate PI facility for consideration. The DDR's will be processed and fulfilled by the TSDR. Data that is not on-line will be requested from the appropriate data archive center.

8.3.3.7 Data Archiving

The TSDR will retain the L-0 through L-4 science products on-line for some period of time after which the data will be sent to the TIMED DAC . Data will later be permanently archived at NSSDC.

8.3.3.8 Science Operations Personnel

The IWG comprises the Project Scientist, Principle Investigators, TIMED Science Operations Coordinator (TSOC), and project theoreticians. The IWG determines the scientific objectives and priorities of the TIMED Mission. The TSOC serves as the liaison between operations (both mission and science) and the IWG. The TSOC also is the administrator of the Long-term Science Plan (LTSP) which describes objectives and priorities

9.0 LAUNCH VEHICLE

Introduction and Purpose

The available Medium Expendable Launch Vehicles (MELV) that will meet the TIMED requirements include the Delta II, the Atlas II and the Titan II. The Delta II has become GSFC's standard MELV. The specific configuration of the recommended Delta II is the 7320 (as shown in Figure 9-1). This vehicle, with 3 graphite epoxy motors (GEM) for boost augmentation, an extended first stage, an uprated second stage and no third stage, is baselined for the TIMED-H launch from the Western Test Range (WTR) and the TIMED-L launch from KSC.

Additional requirements impacting the choice of a launch vehicle were:

- The launch capacity must be sufficient to support the final TIMED design;
- The insertion errors must be within the parameters required for the identified TIMED orbit;
- The inclination from KSC and the Western Space and Missile Center (WSMC) must meet the mission requirements;
- The fairing size must be sufficient for integration; and
- The MELV must be available within the year 2000 timeframe.

The Delta II, two stage configuration was selected since it meets the parameters and outlined objectives; additionally, the TIMED spacecraft fits inside the 9.5 ft. (2.9 m) diameter fairing. The Delta II booster, with its associated solid rocket boosters, provides a minimum margin of 18 percent in its ability to place the two TIMED spacecraft into their required orbits.

9.1 Launch Environment

The launch environment is specific to the launch vehicle which is chosen for the TIMED Mission. It is recognized that lift-off, transonic flight and thrusting will produce the following set of environments which the spacecraft must accommodate. These environments include:

- Steady State Accelerations,
- Quasi-steady State Accelerations,
- Pogo Oscillations,
- Acoustic Noise,
- Random Vibration, and
- Aerodynamic Heating.

The magnitude of these environments will be assessed in Phase B.

9.2 Spacecraft Fairing

The spacecraft is protected by a fairing which shields it from buffeting and aerodynamic heating while in the lower atmosphere. The fairing is jettisoned during second-stage powered flight at a minimum altitude of 67 nmi (125 km).

DELTA II 7320 CONFIGURATION

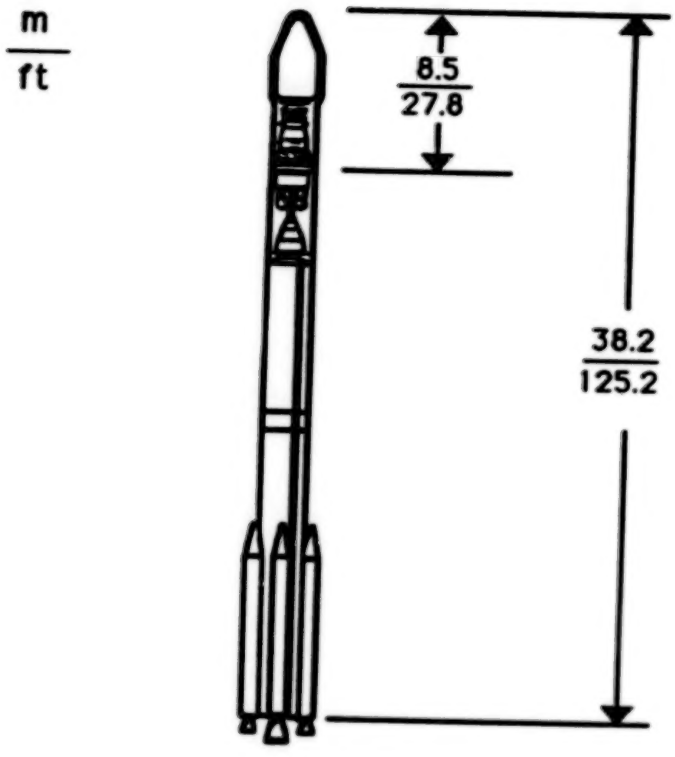


Figure 9-1

Shown in Figure 9-2, the 9.5 ft. Delta fairing is 334.2 inches (8488 mm) long, varies in diameter from 96 inches (2438 mm) at the base to 114 inches (2896 mm) at the widest portion, and weighs approximately 1850 pounds (839 kg). The fairing is an all-aluminum structure fabricated in two half-shells. The half-shells are joined by a contamination-free, linear, piston/cylinder thrusting, separation system that runs longitudinally the full length of the fairing.

The fairing half-shells are jettisoned by actuation of the base and transition separation bolts, and by the detonating fuse in the thrusting joint cylinder rail cavity. Gas pressure generated by the detonating-fuse shears the attach rivets and thrusts the half-shells laterally to attain immediate spacecraft and vehicle clearance. A bellows assembly, within each cylinder rail, retains the detonating-fuse gases to prevent any contamination of the spacecraft during the fairing separation event.

Electrical umbilical cabling provides power to the explosive bolt cartridges, thrusting joint detonators, and the spacecraft (if required), and is attached to the inside surface of the fairing shell. Electrical disconnect is accomplished at fairing separation by quick-disconnect connectors.

Radio frequency (RF) transparent panels can be incorporated into the fairing cylindrical section to permit spacecraft transmission through the fairing. The fiberglass/aluminum foil, biconic section can be made RF-transparent by removal of the aluminum foil.

An air-conditioning inlet door on the fairing provides a controlled environment to the spacecraft and launch vehicle second stage while on the launch stand. A maximum air flow of 1500 ft^3 (42.5 m^3)/min holds the spacecraft temperature within the range of 60 to 80 degrees F (15.5 to 32.2 degrees C) while holding the relative humidity to a maximum of 50 percent.

Contamination of the spacecraft is minimized by frequent cleaning of the fairing surfaces during various stages of the manufacturing process. The surfaces are vacuumed and brushed until all particles visible to the naked eye are removed. All fairing surfaces in the cylindrical and conical sections are sealed to entrap any remaining contamination lodged between them. Capped nutplates are used on all removable fasteners to prevent contamination of the spacecraft during the fairing installation tasks.

A variety of removable doors can be provided in the fairing to permit limited access to the spacecraft following fairing installation. Four standard access doors exist in the baseline fairing configuration. It should be noted that the RF windows and access doors do not contain acoustic or thermal blanket materials. Should the standard doors not satisfy the spacecraft requirements, additional access doors can be provided.

Blockhouse-to-spacecraft electrical wiring terminates in the fairing umbilical cabling which is secured to the inside wall of the fairing.

9.3 Spacecraft Envelope

The allowable static spacecraft envelope within the confines of the fairing are also shown in Figure 9-2 which reflects an envelope for existing attach fittings. The spacecraft envelope defined in these figures represents the maximum allowable spacecraft static dimensions (including manufacturing tolerances) relative to the spacecraft/attach fitting interface. Spacecraft dynamic deflections are taken into account in arriving at envelopes. It is assumed that all significant spacecraft lateral vibration modes are above 15 Hz.

SPACECRAFT ENVELOPE, TWO-STAGE CONFIGURATION (6019 PAF)

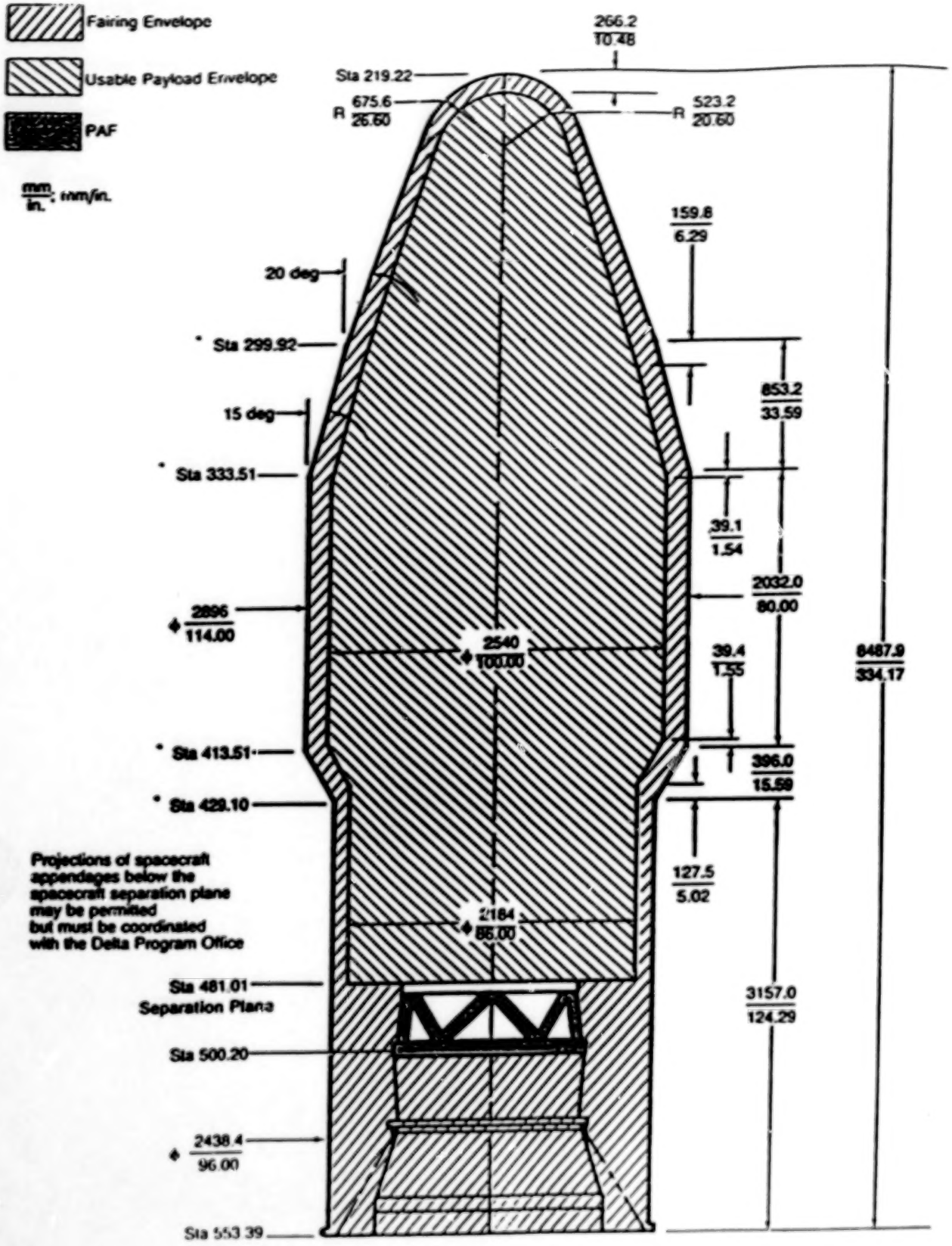


Figure 9-2

In addition to spacecraft dynamic deflections, this envelope includes allowances for vehicle static and dynamic deflections, manufacturing tolerances and acoustic blanket thickness. The TIMED spacecraft, within the launch fairing, is shown in Figure 9-3.

9.4 The 6019 PAF Assembly

The 6019 payload attach fitting (PAF) assembly (shown in Figure 9-4) is approximately 19 inches (483 mm) high and weighs approximate 125 lbs. (57 kg). The one-piece structure is machined from an aluminum forging. The fitting is capable of supporting a spacecraft of 4800 lbs (2177 kg) with a cg of 82 inches (2083 mm) above the separation plane and a radial cg offset of 2 inches (50.8 mm).

The base of the fitting is attached to the forward ring of the second stage. The spacecraft is fastened to the 60 inch (1524 mm) PAF mating diameter at three equally spaced hard points with 0.625 inch (15.9 mm) bolts and explosive actuated nuts. The release of the spacecraft from the launch vehicle occurs when three explosive nuts are simultaneously actuated. The second stage is then backed away by a helium retro (cold gas) system. The separation velocity is approximately 1 ft./sec. (0.3 m/s).

Upon separation, the bolts and catcher assemblies are retained by the spacecraft. Should the TIMED spacecraft require a minimal tip-off rate at separation, a two-step release consisting of the three explosive nuts and a secondary latch system is employed. This necessitates a small lip on the spacecraft at each of the attach bolt locations. Following release of the three explosive nuts, the continued vehicle is stabilized by the launch vehicle attitude control system. Subsequently, three retaining latches are released followed by retrofire of the launch vehicle yielding a minimal separation tip-off of the spacecraft.

SPACECRAFT ENVELOPE, TWO-STAGE CONFIGURATION (6019 PAF)
9.5 FT. FAIRING

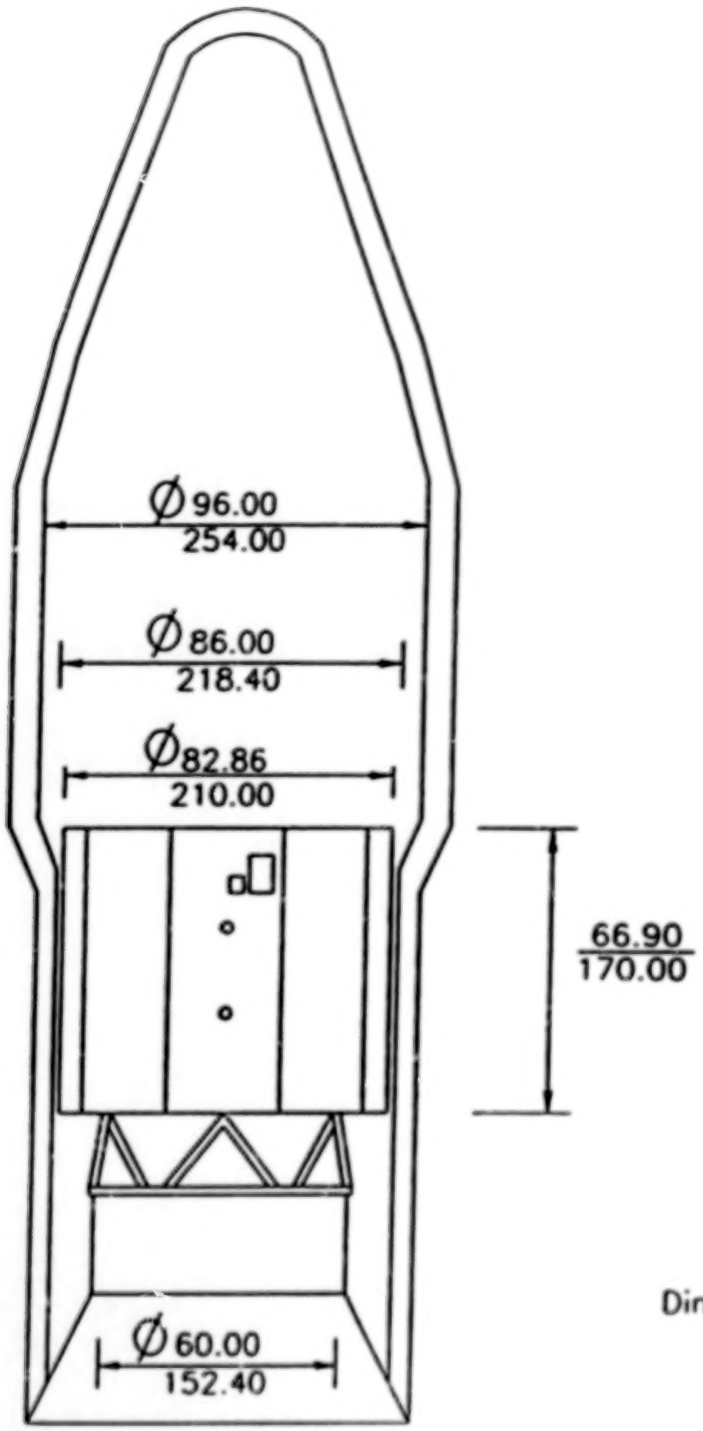


Figure 9-3

TIMED-L WITH PAYLOAD ATTACH FITTING

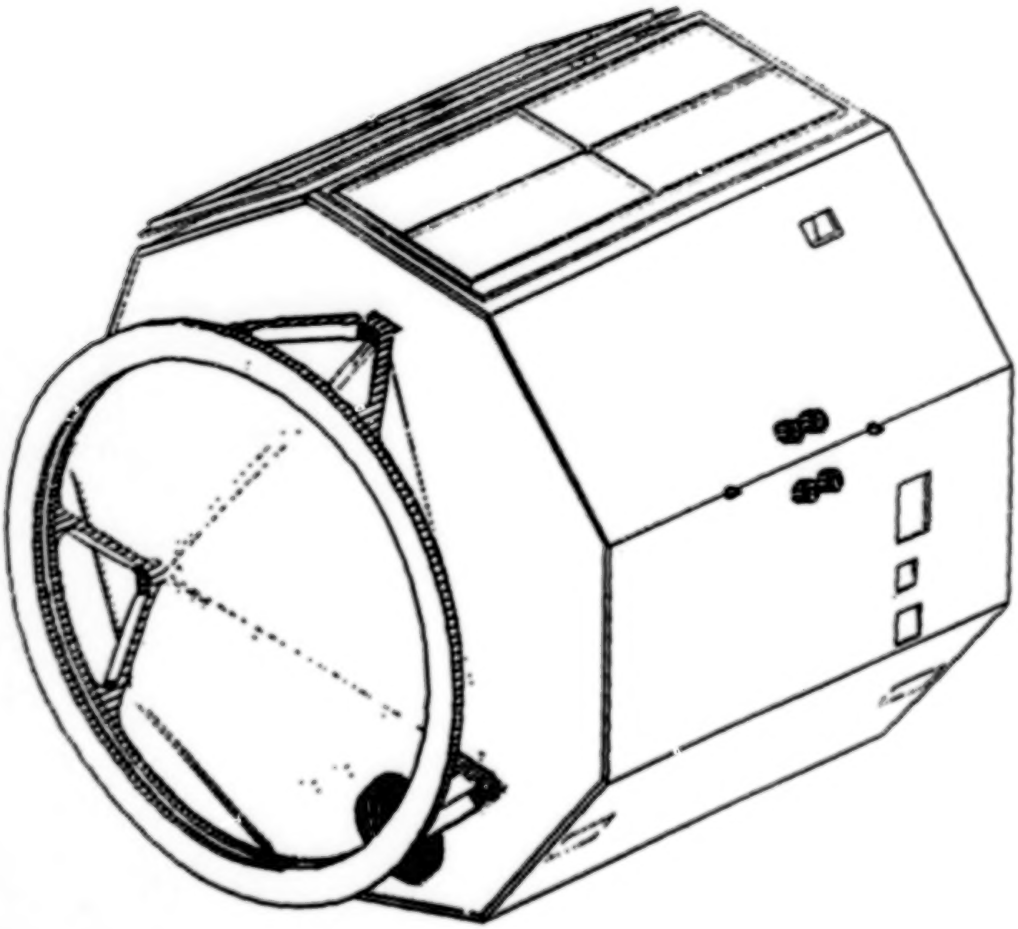


Figure 9-4

APPENDIX A

INSTRUMENT DATA SHEETS

TIMED INSTRUMENT DATA SHEET

DATE: DECEMBER 23, 1991

INSTRUMENT:

FABRY-PEROT INTERFEROMETER

TYPE: REMOTE X IN-SITU _____

SPACECRAFT: H X L X

PURPOSE: VISIBLE/NEAR IR LIMB SCANNER DETECTS DOPPLER SHIFT AND SPREAD OF EMISSION LINES TO DETERMINE WINDS AND TEMPERATURES.

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK 30 WHEN _____

AVERAGE 7 WHEN BELOW 600 KM

STANDBY 3 WHEN ABOVE 600 KM

SURVIVAL _____ WHEN _____

VOLTAGE (V): 28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 40 MINIMUM 0

NON-OPERATING: MAXIMUM 60 MINIMUM -20

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): _____

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):

SERIAL/1.280

NO. & TYPE OF COMMANDS:
26 DISCRETE
126 (32 BIT) SERIAL

TELEMETRY DATA RATE (KBPS):

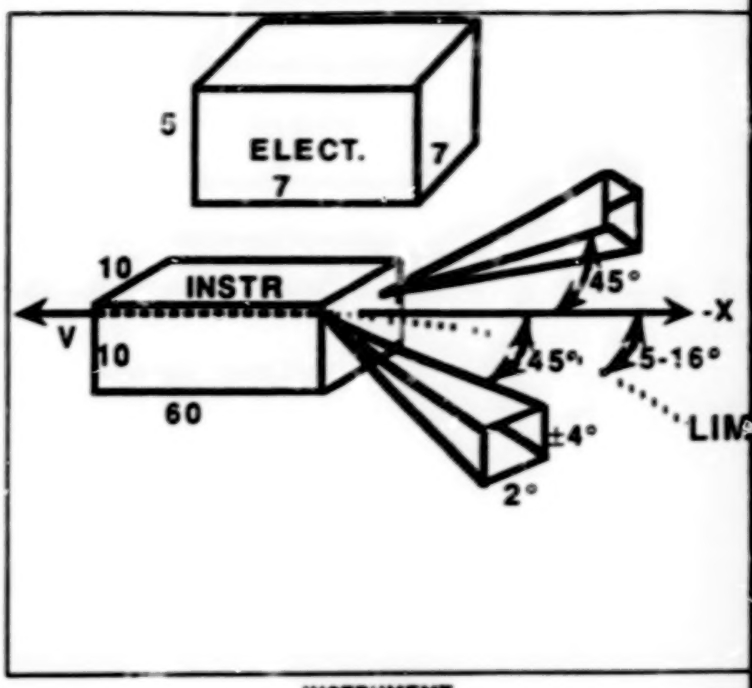
ACS POINTING:

PLACEMENT (°): 1

KNOWLEDGE (°): 0.1

JITTER (°/TIME): 0.1/0.25 SEC

STABILITY (°/TIME): 0.1/16 SEC



INSTRUMENT

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE BELOW 600 KM ALTITUDE.

CIRCULAR ORBIT:

OPERATE CONTINUOUSLY OR AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 2 COMPONENTS

SIZE (cm): L 60/7 W 10/7 H 10/5

FOOTPRINT (cm): L 60/7 W 10/7

MASS (kg): 5+4.6=9.6

S/C SURFACE CUT-OUTS (cm): 10 X 10

FIELD OF VIEW: ±4°V X 2°H (2)

CLEAR VIEW: 45° SEMI-CONE ANGLE

SCAN RATE: 1 SCAN/12 SEC

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

DIRECTION OF VIEW (S): 2 VIEWS ±45° EITHER SIDE OF WAKE, DEPRESSED 5-16°.

CHARGED COUPLED DEVICE (CCD) ARRAY OPERATES AT 200°K. USES TEC TO COOL DETECTORS.

TIMED INSTRUMENT DATA SHEET

DATE: MAY 1, 1992

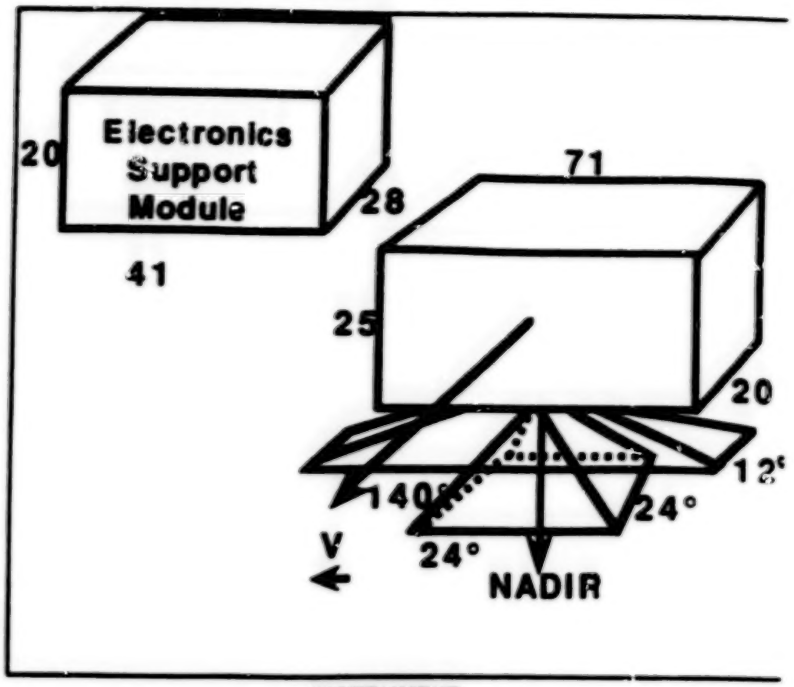
INSTRUMENT:

GLOBAL UV AIRGLOW IMAGER (GUVI)

TYPE: REMOTE X IN-SITU _____

SPACECRAFT: H X L _____

PURPOSE: THE INSTRUMENT IS AN ULTRAVIOLET SCANNING SPECTROGRAPH WHICH IMAGES THE ATMOSPHERE IN THE 1200-3000 Å SPECTRAL RANGE. IT MONITORS THE EXCHANGE OF CHEMICAL SPECIES, WAVE MOMENTUM STRESSES, AND PARTICLE AND WAVE ENERGY BETWEEN THE MESOSPHERE AND THERMOSPHERE.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK	35	WHEN	25% OF NORMAL OPERATION
AVERAGE	30	WHEN	75% OF NORMAL OPERATION
STANDBY	5	WHEN	0%
SURVIVAL		WHEN	

VOLTAGE (V): 28 ± 1.4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 35 MINIMUM -15

NON-OPERATING: MAXIMUM 50 MINIMUM -30

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): 5

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
11

NO. & TYPE OF COMMANDS:
256, 8-BIT

TELEMETRY DATA RATE (KBPS):
TBD

ACS POINTING:

PLACEMENT (°): 1

KNOWLEDGE (°): 0.05

JITTER (°/TIME): 0.05°/0.1 SEC

STABILITY (°/TIME): 0.1°/10 SEC

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE AS MUCH AS DUTY CYCLING PERMITS.

CIRCULAR ORBIT:

OPERATE AS MUCH AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 2 COMPONENTS
ELECTRONICS/ELECT. SUPPORT MODULE

SIZE (cm): L 71/41 W 20/28 H 25/20

FOOTPRINT (cm): L 71/41 W 20/28

MASS (kg): 20

S/C SURFACE CUT-OUTS (cm): 15.2 CM DIA. CIRCLE
13.5 X 30.5
12° ALONG TRACK

FIELD OF VIEW: 140° CROSS TRACK, 24° X 24°

CLEAR VIEW: 20° BORDER

SCAN RATE: _____

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

THE INSTRUMENT LOOKS IN THE NADIR DIRECTION

CONTAMINATION BELOW 200 KM IS A CONCERN.

13.5 X 30.5 CM S/C SURFACE CUT-OUT ASSUMES INSTRUMENT SENSOR RECESSED 15.2 CM INSIDE S/C SURFACE.

TIMED INSTRUMENT DATA SHEET

DATE: DECEMBER 23, 1991

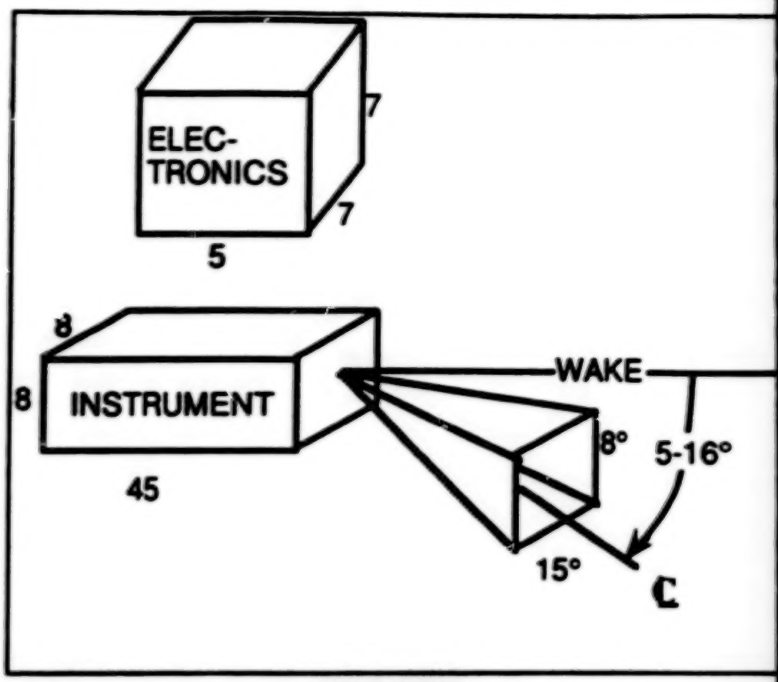
INSTRUMENT:

IMAGING PHOTOMETER

TYPE: REMOTE X IN-SITU

SPACECRAFT: H X L X

PURPOSE: VISIBLE/NEAR IR LIMB SCANNER DETECTS VARIOUS FEATURES IN THE AIRGLOW AND AURORA AND ENABLES GRAVITY WAVE PARAMETERS TO BE DERIVED.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W): STEPPER MOTOR

PEAK 30 WHEN TIMES 300 ms

AVERAGE 10 WHEN BELOW 900 KM

STANDBY 4 WHEN ABOVE 900 KM

SURVIVAL WHEN

VOLTAGE (V): 28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C).

OPERATING: MAXIMUM 40 MINIMUM 0

NON-OPERATING: MAXIMUM 60 MINIMUM -20

SURVIVAL: MAXIMUM MINIMUM

HEATER POWER (W):

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
SERIAL/1.280

NO. & TYPE OF COMMANDS:
50/ORBIT SERIAL

TELEMETRY DATA RATE (KBPS):

ACS POINTING:

PLACEMENT (°): 1

KNOWLEDGE (°): 0.1

JITTER (°/TIME): 0.1/0.25 SEC

STABILITY (°/TIME): 0.1/16 SEC

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE BELOW 900 KM ALTITUDE.

CIRCULAR ORBIT:

OPERATE CONTINUOUSLY OR AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 2 COMPONENTS

SIZE (cm): L 45/7 W 8/7 H 8/5

FOOTPRINT (cm): L 45/7 W 8/5

MASS (kg): 5+3.4=8.4

S/C SURFACE CUT-OUTS (cm): 8 X 8

FIELD OF VIEW: ±4°V X ±7.5°H

CLEAR VIEW: 45° SEMI-CONE ANGLE

SCAN RATE:

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

DIRECTION OF VIEW: LIM ; WAKE, DEPRESSED 5-16°.

CHARGED COUPLED DEVICE (CCD) (512X512) ARRAY OPERATES AT 200°K. USES TEC TO COOL DETECTORS.

SCIENCE SAMPLE RATE ~1/4 SEC.

TIMED INSTRUMENT DATA SHEET

DATE: JANUARY 16, 1992

INSTRUMENT:

INFRARED LIMB SOUNDER (IRLS)

TYPE: REMOTE ___X___ IN-SITU_____

SPACECRAFT: H_____ L___X___

PURPOSE: THE INSTRUMENT IS A CRYOGENICALLY COOLED LIMB SCANNING INSTRUMENT THAT LOOKS IN THE OFF-WAKE DIRECTION AND MAKES GLOBAL MEASUREMENTS OF KINETIC TEMPERATURE, OZONE, NO, NO_2, H_2O, AND OH EMISSIONS.

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W): START-UP 2 COOLERS: 20 MSEC PULSES
PEAK 220 WHEN FOR ≤10 MINS

AVERAGE 115 WHEN 100%

STANDBY 44 WHEN_____
HEATERS

SURVIVAL 44 WHEN_____

VOLTAGE (V): 28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 100 CONDUCTIVE 0

TEMPERATURE (°C): RADIATOR A/B
OPERATING: MAXIMUM 15/-5 MINIMUM 5/-13
RADIATOR A/B
NON-OPERATING: MAXIMUM 125/85 MINIMUM -55/-45
RADIATOR A/B
SURVIVAL: MAXIMUM 125/85 MINIMUM -55/-45

HEATER POWER (W): 44

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
1553 BUS, 3 KBPS

NO. & TYPE OF COMMANDS:
27 DISCRETES, 24 SERIAL

TELEMETRY DATA RATE (KBPS):
1553 BUS, 500 BPS

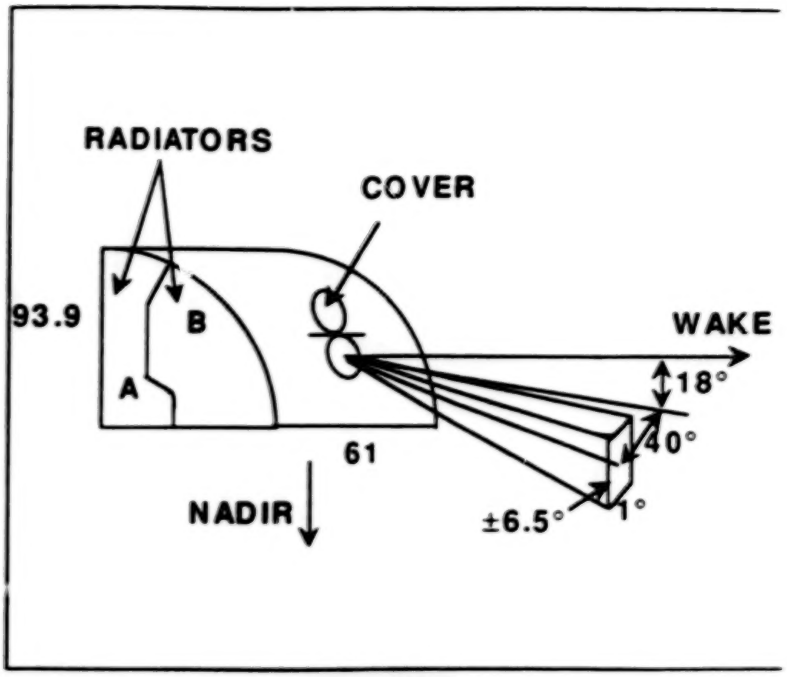
ACS POINTING:

PLACEMENT (°): 1.0

KNOWLEDGE (°): 0.1° PITCH & YAW, 0.25° ROLL

JITTER (°/TIME): ≤ 20 ARC-SEC/SEC

STABILITY (°/TIME): ≤0.0075°/18 SEC



INSTRUMENT

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

USEFUL SCIENTIFIC DATA CAN BE OBTAINED FOR ORBIT ALTITUDES BETWEEN 1000 KM AND 300 KM.

CIRCULAR ORBIT:

OPERATE 100% AT 500 KM (450 KM ACCEPTABLE) OR AS MUCH AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT
93.9 RADIUS X 90° ARC X 61 WIDE

SIZE (cm): L_____ W 61 H_____

FOOTPRINT (cm): L 93.9 W 61

MASS (kg): 72

S/C SURFACE CUT-OUTS (cm): 10 X 20 ELLIPSE
40° OFF WAKE, DEPRESSED 18°, SHADY SIDE

FIELD OF VIEW: ±6.5° V X 1° H

CLEAR VIEW: 35° V X 25° H (SEMI-CONE ANGLES)

SCAN RATE: 0.25°/SEC

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

THE INSTRUMENT LOOKS AT THE EARTH'S LIMB 40° OFF THE WAKE DIRECTION, DEPRESSED 18°, ON THE SHADY SIDE.
HAS TWO STIRLING-CYCLE COOLERS TO COOL THE DETECTOR ARRAY TO 75° K.
REQUIRES TWO THERMAL CONTROL RADIATOR SURFACE AREAS (0.49 + 0.2 M^2)TOTALING 0.69 M^2 IN ANTI-SOLAR DIRECTION.

TIMED INSTRUMENT DATA SHEET

DATE: JANUARY 7, 1992

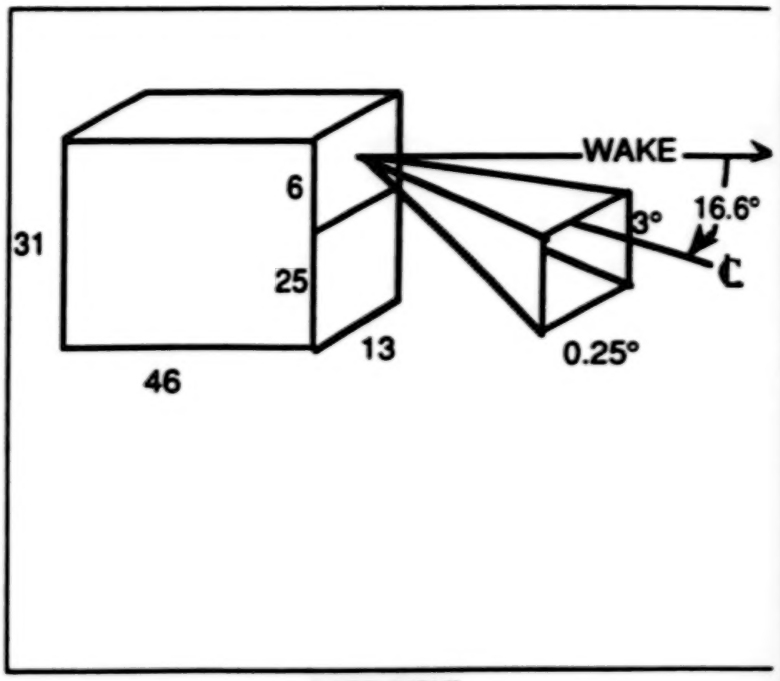
INSTRUMENT:

NEAR INFRARED SPECTROMETER/PHOTOMETER (NIRS/P)

TYPE: REMOTE ___X___ IN-SITU _____

SPACECRAFT: H ___X___ L _____

PURPOSE: THE INSTRUMENT EMPLOYS AN ARRAY DETECTOR TO PROVIDE SCANNING IN BOTH WAVELENGTH AND SPATIAL DIRECTIONS AND MEASURES THE O_2 EMISSION AT 1.27 μm IN THE SINGLE DELTA EXCITED STATE. THE OZONE DENSITY CAN BE INFERRED FROM THE ALTITUDE DEPENDENCE OF THE EMISSION.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK ___9.5___ WHEN _____

AVERAGE ___7.5___ WHEN ___100%___

STANDBY _____ WHEN _____

SURVIVAL _____ WHEN _____

VOLTAGE (V): ___-28___ ± ___4___

THERMAL:

DISSIPATION (%): RADIATIVE ___20___ CONDUCTIVE ___80___

TEMPERATURE (°C):

OPERATING: MAXIMUM ___40___ MINIMUM ___0___

NON-OPERATING: MAXIMUM ___60___ MINIMUM ___-20___

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): ___HEATER, WHEN NEEDED___

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
16-BIT SERIAL WORDS, 256 BITS/SEC

NO. & TYPE OF COMMANDS:
16 SERIAL, ≤ 256 BITS/ORBIT

TELEMETRY DATA RATE (KBPS):

ACS POINTING:

PLACEMENT (°): ___1.0___

KNOWLEDGE (°): ___0.1___

JITTER (°/TIME): ___0.007/0.03 SEC___

STABILITY (°/TIME): ___20 ARC-SEC/12 SEC___

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE 100% AT ≤ 1500 KM ALTITUDE OR AS MUCH AS DUTY CYCLING PERMITS.

CIRCULAR ORBIT:

OPERATE 100% OR AS MUCH AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT

SIZE (cm): L ___46___ W ___13___ H ___31___

FOOTPRINT (cm): L ___46___ W ___13___

MASS (kg): ___5___

S/C SURFACE CUT-OUTS (cm): ___13 X 6___

FIELD OF VIEW: ___3° V X 0.25° H___

CLEAR VIEW: ___22.5° SEMI-CONE ANGLE___

SCAN RATE: ___1 SCAN/4 SEC___

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)
LOOKS 16.6° BELOW WAKE DIRECTION AT EARTH'S LIMB.

INSTRUMENT HAS 2 X 32 GERMANIUM PHOTODIODE ARRAY.

REQUIRES 0.02 M^2 RADIATOR TO DISSIPATE 2.5 W.

ALIGN WITH UVS.

TIMED INSTRUMENT DATA SHEET

DATE: MARCH 1, 1992

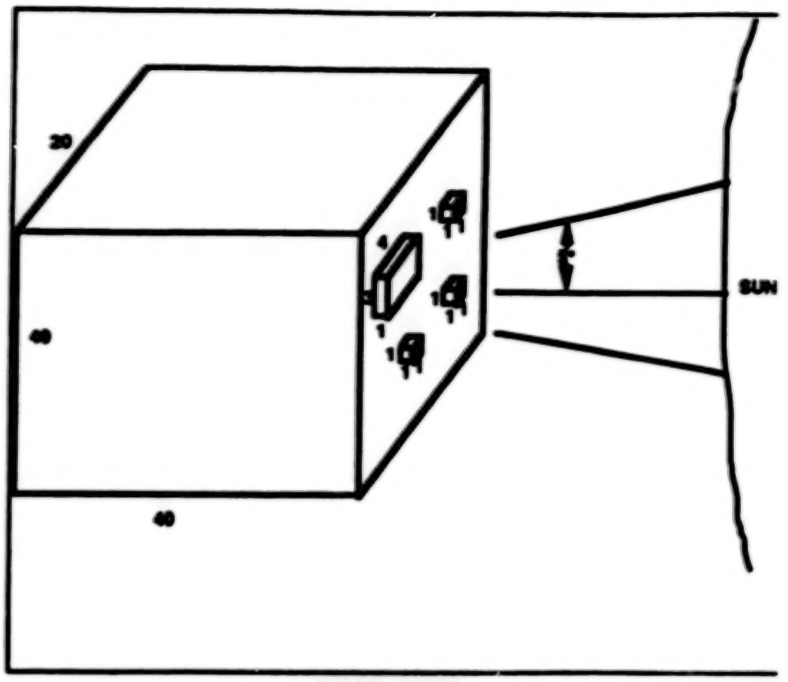
INSTRUMENT:

SOLAR EUV SPECTROMETER (SEUVS)

TYPE: REMOTE X IN-SITU _____

SPACECRAFT: I X L _____

PURPOSE: INSTRUMENT IS MOUNTED ON A SOLAR POINTING PLATFORM (SPP) AND MEASURES THE SOLAR SPECTRAL IRRADIANCE OVER THE WAVE-LENGTH RANGE FROM 50 TO 2000 Å.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK 14 WHEN ≤10 MIN/ORBIT

AVERAGE 7.5 WHEN DURING 10 MIN OP.

STANDBY 7 OR 0 WHEN CAN BE OFF

SURVIVAL _____ WHEN _____

VOLTAGE (V): 28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 30 MINIMUM 0

NON-OPERATING: MAXIMUM 70 MINIMUM -50

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): <10

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
8-BIT SERIAL DIGITAL, AVERAGE RATE: 0.4/10 MIN, PEAK RATE: 17.

NO. & TYPE OF COMMANDS:
16 SINGLE-BIT, DIGITAL

TELEMETRY DATA RATE (KBPS):

ACS POINTING:

PLACEMENT (°): 0.5°

KNOWLEDGE (°): 0.1°

JITTER (°/TIME): 1 ARC-MIN/SEC

STABILITY (°/TIME): _____

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE ~10 MIN/ORBIT (MINIMUM 2 MIN/ORBIT) ABOVE 500 KM POINTING AT SUN.

CIRCULAR ORBIT:

OPERATE ~10 MIN/ORBIT (MINIMUM 2 MIN/ORBIT) ANYWHERE IN ORBIT POINTING AT SUN.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT

SIZE (cm): L 40 W 20 H 40

FOOTPRINT (cm): L 40 W 20

MASS (kg): 11+7=18

S/C SURFACE CUT-OUTS (cm): 20 X 40

FIELD OF VIEW: 5° SEMI-CONE ANGLE

CLEAR VIEW: 7.5° SEMI-CONE ANGLE

SCAN RATE: _____

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

MOUNTED ON SPP ON SUN SIDE OF S/C AND POINTED AT SUN.

HAS 3 APERTURE DOORS (4 X 3, 1X1, 1X1 CM), CLOSED LOOP SUN TRACKER, AND MCP-ARRAY DETECTORS (SOMEWHAT SENSITIVE TO S/C MAGNETIC FIELD).

PREFERS NEAR CONSTANT TEMPERATURE.

TIMED INSTRUMENT DATA SHEET

DATE: MAY 1, 1992

INSTRUMENT:

UV SPECTROMETER (UVS)

TYPE: REMOTE X IN-SITU

SPACECRAFT: H X L

PURPOSE: INSTRUMENT MEASURES THE ALTITUDE DEPENDENCE OF RAYLEIGH SCATTERING IN THE WAVELENGTH RANGE OF 250-340 nm. IT HAS A MOVABLE GRATING AND A SCAN MIRROR TO VIEW THE EARTH'S LIMB. OZONE ABUNDANCE IS DERIVED, AND NEUTRAL TEMPERATURES ARE DETERMINED.

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W): SCAN & TURN-ON

PEAK 7 WHEN MSEC

AVERAGE 5 WHEN 67% (DAY OPS.)

STANDBY WHEN

SURVIVAL WHEN

VOLTAGE (V): -28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 40 MINIMUM 0

NON-OPERATING: MAXIMUM 60 MINIMUM -20

SURVIVAL: MAXIMUM MINIMUM

HEATER POWER (W): 4

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
12-BIT SERIAL WORDS, 64 BITS/SEC

NO. & TYPE OF COMMANDS:
16 SERIAL, ≤ 256 BITS/ORBIT

TELEMETRY DATA RATE (KBPS):

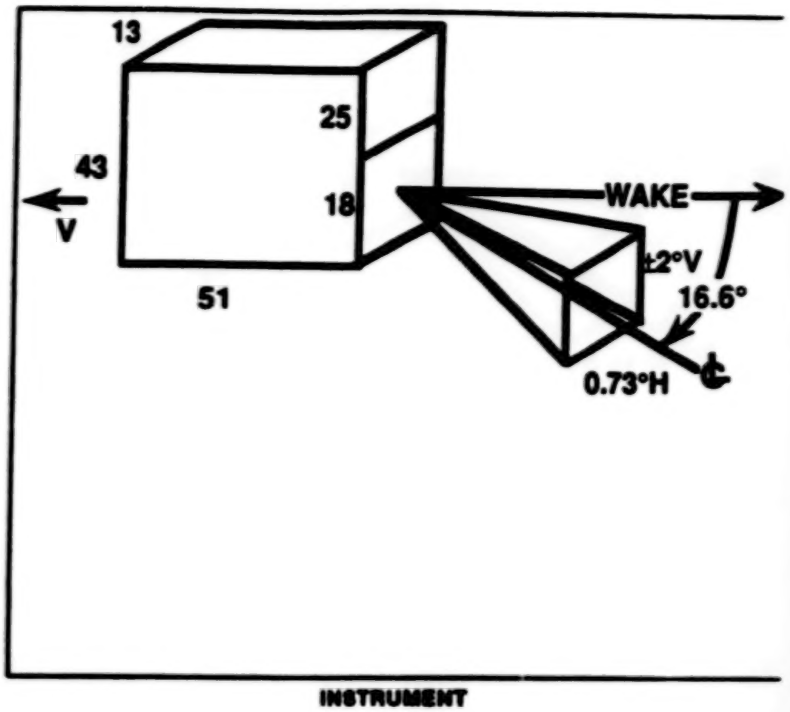
ACS POINTING:

PLACEMENT (°): 1.0

KNOWLEDGE (°): 0.1

JITTER (°/TIME): 0.007/0.03 SEC

STABILITY (°/TIME): 30 ARC-SEC/12 SEC



INSTRUMENT

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE AT ≤ 1500 KM ALTITUDE DURING DAYLIGHT. DURING NIGHT: SOME CALIBRATION, OFF OTHERWISE.

CIRCULAR ORBIT:

OPERATE DURING DAYLIGHT. DURING NIGHT: SOME CALIBRATION, OFF OTHERWISE.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT

SIZE (cm): L 51 W 13 H 43

FOOTPRINT (cm): L 51 W 13

MASS (kg): 7.3

S/C SURFACE CUT-OUTS (cm): 13 X 18

FIELD OF VIEW: ±2° V X 0.73° H

CLEAR VIEW: 25° SEMI-CONE ANGLE

SCAN RATE: 1 SCAN/12 SEC

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

LOOKS 16.6° BELOW WAKE DIRECTION AT EARTH'S LIMB.

CONTAINS DUAL PHOTOMULTIPLIER TUBES.

ALIGN WITH NIRS.

TIMED INSTRUMENT DATA SHEET

DATE: DECEMBER 30, 1991

INSTRUMENT:

ION DRIFT METER/RETARDING POTENTIAL ANALYZER (IDM/RPA)

TYPE: REMOTE _____ IN-SITU X

SPACECRAFT: H X L X

PURPOSE: THE IDM/RPA LOOKS IN THE RAM DIRECTION. THE IDM MEASURES THE TRANSVERSE COMPONENTS OF ION VELOCITY. THE RPA MEASUREMENTS RESULT IN DERIVATIONS OF ION DENSITIES, TEMPERATURES, AND THE COMPONENT OF ION DRIFT VELOCITY INTO THE SENSOR.

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK 8 WHEN _____

AVERAGE 8 WHEN ON

STANDBY 0.5 WHEN OFF

SURVIVAL _____ WHEN _____

VOLTAGE (V): -28 ± 4

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 50 MINIMUM -20

NON-OPERATING: MAXIMUM 60 MINIMUM -30

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): NOT REQUIRED

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
2.2

NO. & TYPE OF COMMANDS:
1 WORD, SERIAL

TELEMETRY DATA RATE (KBPS):

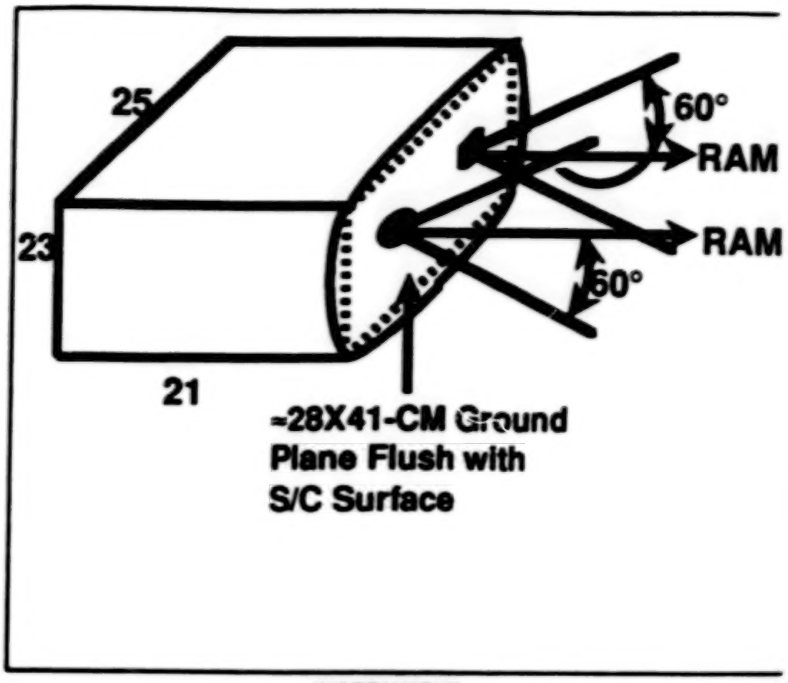
ACS POINTING:

PLACEMENT (°): 1

KNOWLEDGE (°): 0.1

JITTER (°/TIME): 0.5/SEC

STABILITY (°/TIME): _____



INSTRUMENT

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE CONTINUOUSLY AT ALL ALTITUDES OR AS MUCH AS DUTY CYCLING PERMITS.

CIRCULAR ORBIT:

OPERATE CONTINUOUSLY OR AS MUCH AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT

SIZE (cm): L 21 W 25 H 23

FOOTPRINT (cm): L 21 W 25

MASS (kg): 8

S/C SURFACE CUT-OUTS (cm): ~28 X 41

FIELD OF VIEW: 60° SEMI-CONE ANGLE

CLEAR VIEW: 60° SEMI-CONE ANGLE

SCAN RATE: _____

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

THE INSTRUMENT HAS TWO PLANAR SENSORS LOOKING IN THE RAM DIRECTION.
REQUIRES GROUND PLANE ~28 X 41 cm FLUSH WITH SPACECRAFT EXTERIOR SURFACE, CONDUCTIVE S/C, AND ISOLATED SOLAR ARRAY.
REQUIRES UNOBSTRUCTED VIEW IN RAM DIRECTION. ALIGN WITH NMS/WT.
REQUIRES S/C MAGNETIC FIELD ≤ 0.01 GAUSS AT SENSOR APERTURES.
PREFERS S/C POWER SYSTEM TO HAVE POSITIVE GROUND.

TIMED INSTRUMENT DATA SHEET

DATE: DECEMBER 30, 1991

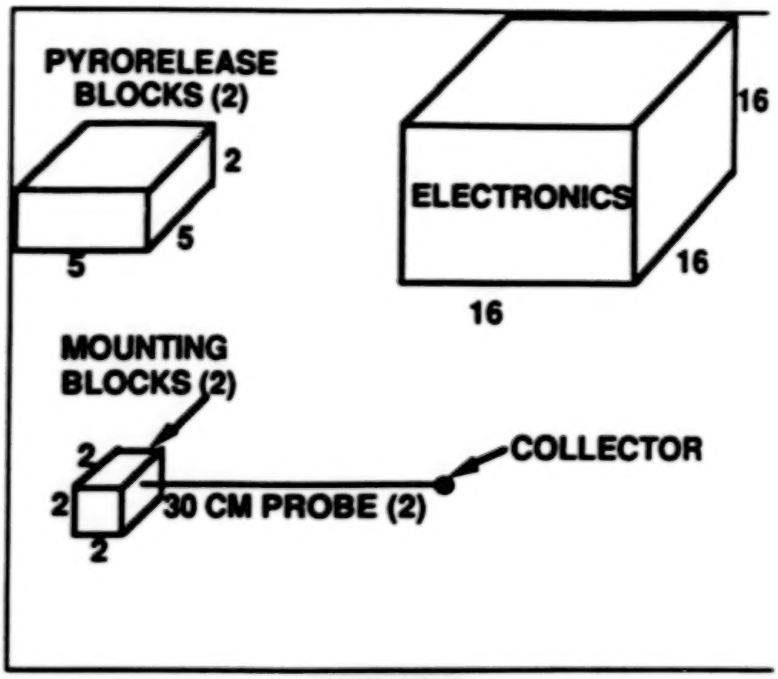
INSTRUMENT:

LANGMUIR PROBE (LP)

TYPE: REMOTE _____ IN-SITU X

SPACECRAFT: H X L X

PURPOSE: COLLECTORS MOUNTED ON THE ENDS OF TWO SHORT BOOMS MEASURE THE ELECTRON TEMPERATURE AND THE ION AND ELECTRON DENSITY.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK 4 WHEN 0

AVERAGE 4 WHEN BELOW 1000 KM

STANDBY 0 WHEN ABOVE 1000 KM

SURVIVAL _____ WHEN _____

VOLTAGE (V): 28 ± 2

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 50 MINIMUM -10

NON-OPERATING: MAXIMUM 60 MINIMUM -10

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): N/A

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
1.0

NO. & TYPE OF COMMANDS:

TELEMETRY DATA RATE (KBPS):

ACS POINTING:

PLACEMENT (°): 5

KNOWLEDGE (°): 1

JITTER (°/TIME): 1/ SEC

STABILITY (°/TIME): 1/ SEC

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE WITH OTHER IN-SITU INSTRUMENTS BELOW 1000 KM ALTITUDE.

CIRCULAR ORBIT:

OPERATE CONTINUOUSLY WITH OTHER IN-SITU INSTRUMENTS OR AS DUTY CYCLING PERMITS.

INSTRUMENT CHARACTERISTICS: 7 COMPONENTS
TWO PROBES 0.4 cm DIAM. X 30 cm LONG

SIZE (cm): L 16/5/2 W 16/5/2 H 16/2/2

FOOTPRINT (cm): L 16/5/2 W 16/5/2

MASS (kg): 1.7+0.4+0.3+0.1=2.5

S/C SURFACE CUT-OUTS (cm): 5/2 X 5/2
DETECTORS ON OUTER 5 cm OF PROBES

FIELD OF VIEW: 30° SEMI-CONE ANGLE ABOUT RAM

CLEAR VIEW: 90° SEMI-CONE ANGLE ABOUT RAM

SCAN RATE: _____

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

TWO DEPLOYABLE PROBES, PYROTECHNICALLY RELEASED.

PROBES LOCATED PERPENDICULAR TO VELOCITY VECTOR, PREFERABLY ONE PARALLEL TO ZENITH DIRECTION AND ONE PARALLEL TO NADIR DIRECTION.

TIMED INSTRUMENT DATA SHEET

DATE: DECEMBER 31, 1991

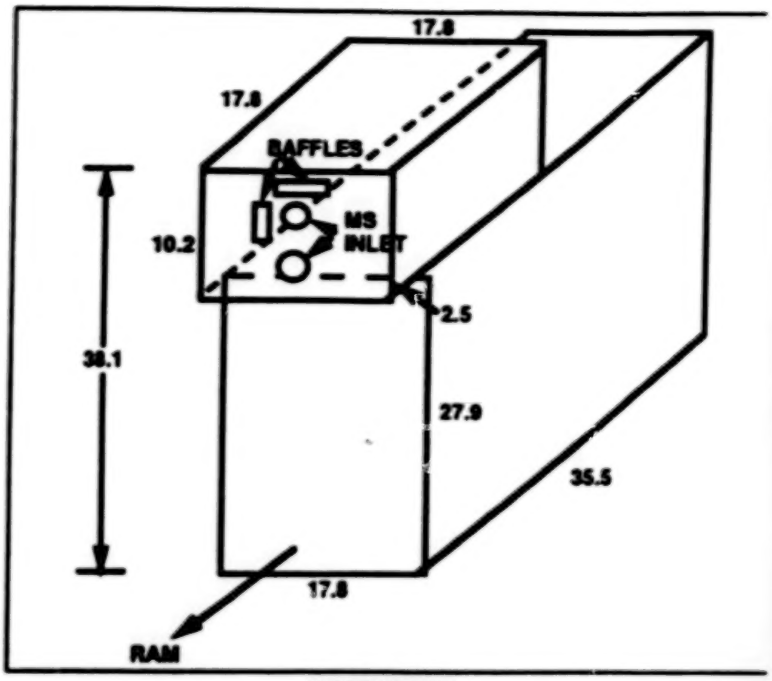
INSTRUMENT:

NEUTRAL MASS SPECTROMETER/WIND AND TEMPERATURE (NMS/WT)

TYPE: REMOTE _____ IN-SITU X

SPACECRAFT: H X L X

PURPOSE: THE INSTRUMENT LOOKS INTO THE RAM AND MEASURES THE COMPOSITION OF THE THERMOSPHERIC NEUTRAL CONSTITUENTS AND THE COMPONENTS OF WIND. IT HAS MOVABLE BAFFLES AND WILL EMPLOY EITHER MAGNETIC OR RF SEPARATION OF THE IONS OF DIFFERENT MASSES.



INSTRUMENT

INSTRUMENT RESOURCE REQUIREMENTS:

POWER (W):

PEAK 30 WHEN _____

AVERAGE 24 WHEN BELOW 600 KM

STANDBY 1-2 WHEN ABOVE 600 KM

SURVIVAL _____ WHEN _____

VOLTAGE (V): 28 ± 6

THERMAL:

DISSIPATION (%): RADIATIVE 20 CONDUCTIVE 80

TEMPERATURE (°C):

OPERATING: MAXIMUM 40 MINIMUM -20

NON-OPERATING: MAXIMUM 60 MINIMUM -20

SURVIVAL: MAXIMUM _____ MINIMUM _____

HEATER POWER (W): _____

C&DH:

SCIENCE DATA FORMAT & RATE (KBPS):
1.536

NO. & TYPE OF COMMANDS:

TELEMETRY DATA RATE (KBPS):

ACS POINTING:

PLACEMENT (°): 2.0

KNOWLEDGE (°): 0.05

JITTER (°/TIME): 0.05/MIN

STABILITY (°/TIME): 0.05/MIN

OPERATIONAL SCENARIO:

ELLIPTICAL ORBIT:

OPERATE BELOW 600 KM. STANDBY MODE ABOVE 600 KM.

CIRCULAR ORBIT:

OPERATE BELOW 600 KM. STANDBY MODE ABOVE 600 KM.

INSTRUMENT CHARACTERISTICS: 1 COMPONENT

SIZE (cm): L 35.5 W 17.8 H 38.1

FOOTPRINT (cm): L 35.5 W 27.9

MASS (kg): 12

S/C SURFACE CUT-OUTS (cm): 17.8 X 10.2

FIELD OF VIEW: ±90° ABOUT RAM

CLEAR VIEW: ±90° ABOUT RAM

SCAN RATE: _____

UNIQUE REQUIREMENTS/CHARACTERISTICS:
(e.g. BOOMS, COVERS, ENVIRONMENTAL, ETC.)

METAL CERAMIC BREAK-OFF COVER IS PYROTECHNICALLY ACTUATED AND SEPARATED IN ORBIT.

HAS TWO MOVABLE BAFFLES WHICH MOVE OVER ONE SENSOR INLET.

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APPENDIX B: GLOSSARY

Metadata is defined as information about data sets which provides a description of the content, format and utility of the data sets.

Processing Levels:

Level-0	Time-ordered, error-detected, duplicate packet removed raw science counts.
Level-1	Calibrated science data, packaged with needed ancillary and engineering data.
Level-2	Level-1 data converted to geophysical parameters.
Level-3	Level-2 data mapped on uniform grids.
Level-4	Model outputs or results from analysis of lower level data.

APPENDIX C: ACRONYM LIST

ACC	Accelerometer
ACE	Attitude Control Electronics
ACS	Attitude Control System
AE	Atmospheric Explorer
AMAO	Advanced Missions Analysis Office
AOS	Advanced Orbiting Systems
ASIC	Application Specific Integrated Circuit
BIU	Bus Interface Unit
BOL	Beginning of Life
CCD	Charge Coupled Device
CCSDS	Consultative Committee for Space Data Systems
CCTV	Closed Circuit Television
cg	Center-of-Gravity
C&DH	Command and Data Handling
CMF	Command Management Facility
CMS	Command Management Services
COBE	Cosmic Background Explorer
COTS	Commercial Off the Shelf
CPU	Central Processing Unit
DAC	Data Archive Center
DAR	Data Acquisition Request
DE	Dynamics Explorer
DET	Direct Energy Transfer
DCF	Data Capture Facility
DDR	Data Distribution Requests
DOD	Department of Defense
DMA	Direct Memory Access
DSN	Deep Space Network
EED	Electro-Explosive Device
EEPROM	Electronically Erasable Programmable Read Only Memory
EDAC	Error Dectection and Correction
EGSE	Electrical Ground Support Equipment
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
EOS	Earth Observing System
EOSDIS	Earth Observing System Data Information System
EPS	Electrical Power Subsystem
ERBE	Earth Radiation Budget Experiment
EUV	Extreme Ultraviolet
EUVE	Extreme Ultraviolete Explorer
FDF	Flight Dynamics Facility
FEM	Finite Element Model
FEU	Flight Equivalent Unit
FOT	Flight Operations Team
FOV	Field of View
FPI	Fabry-Perot Interferometer
Gb or GB	Gigabytes
GEM	Graphite Epoxy Motors
GGS	Global Geospace Science
GN	Ground Network
GSFC	Goddard Space Flight Center

APPENDIX C: ACRONYM LIST (CONTD.)

GSTDN	Ground Station Tracking and Data Network
GRO	Gamma Ray Observatory
GUVI	Global Ultraviolet Imager
IDM/RPA	Ion Drift Meter/Retarding Potential Analyzer
IMS	Ion Mass Spectrometer
IP	Imaging Photometer
IRLS	Infrared Limb Sounder
ISTP	International Solar Terrestrial Physics
I&T	Integration and Test
IWG	Investigator's Working Group
JPL	Jet Propulsion Laboratory
Kb	Kilobits
KSC	Kennedy Space Center
LP	Langmuir Probe
LTSP	Long Term Science Plan
LTOP	Long Term Operations Plan
LVPS	Low Voltage Power Supply
LZP	Level Zero Processing
Mb	Megabits
MCP	Microchannel Plate
MEV	Million Electron Volts
MELV	Medium Expendible Launch Vehicle
MGSE	Mechanical Ground Support Equipment
MLI	Multi-layer Insulation
MLTI	Mesosphere and Lower-Thermosphere/Ionosphere
MOCC	Mission Operations Control Center
MOP	Mission Operations Plan
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications
NCC	Network Control Center
NGT	NASA Ground Terminal
NIRS/P	Near Infrared Spectrometer/Photometer
NMS	Newton Meter Second
NMSWT	Neutral Mass Spectrometer/Wind and Temperature
NOAA	National Oceanic and Atmospheric Administration
NSI	NASA Science Internet
OGO	Orbiting Geophysical Observatory
PAF	Payload Attach Fitting
PI	Principal Investigator
PROM	Programmable Read Only Memory
PSE	Power Supply Electronics
QL	Quick-look
RAO	Resource Analysis Office
RF	Radio Frequency
ROM	Read Only Memory
rpo	Revolutions per Orbit
rpm	Revolutions per Minute
SAMPEX	Small Atmospheric Explorer
SDF	Software Development Facility
SDT	Science Definition Team
SEUV	Solar Extreme Ultraviolet
SEUVS	Solar Extreme Ultraviolet Spectrometer

APPENDIX C: ACRONYM LIST (CONTD.)

SINDA	A Thermal Analyses Modeling Program
SLOC	Source Lines of Code
SME or SMEX	Solar Mesospheric Explorer
SPP	Solar Pointing Platform
STDN	Space Tracking Data Network
STIP	Short Term Instrument Plan
STOP	Short Term Operations Plan
SWG	Science Working Group
TB	Terrabytes
TCS	Thermal Control System
TDF	TIMED Development Facility
TDRSS	Tracking and Data Relay Satellite System
TEC	Thermal Electric Cooler
TIMED	Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics
TIMED-H	Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics - High Inclination
TIMED-L	Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics - Low Inclination
TOO	Targets-of-Opportunity
TOTS	Transportable Orbital Tracking System
TRMM	Tropical Rainfall Measuring Mission
TSDR	TIMED Science Data Room
TSOC	TIMED Science Operations Coordinator
TSDR	TIMED Science Data Room
UARS	Upper Atmosphere Research Satellite
UV	Ultraviolet
UVS	Ultraviolet Spectrometer
VRTX	Virtual Realtime Executive
VSS	Voice Switching System
WFF	Wallops Flight Facility
WOTS	Wallops Orbital Tracking Station
WSC	White Sands Complex
WTR	Western Test Range
XTE	X-Ray Timing Explorer

APPENDIX D: REFERENCE DOCUMENTS

1. Space Science in the Twenty-First Century: Imperatives for the Decades 1995-2015, Space Science Board, National Academy of Sciences, 1988.

2. The TIMED Mission and Science Program, Report of the Science Definition Team, Volumes I, II and III, July 1991, Revised January 1992.

3. TIMED Spacecraft Pre-Phase A Study Report, Revision A, June 1991.

4. TIMED Mission Announcement of Opportunity, AO No. 92-OSSA-1, April 21, 1992.

5. Phase A Study Report for TIMED Ground Systems, July 1992.

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7. Commercial Delta II Payload Planners Guide, MDCH 3224B.

• U.S. GOVERNMENT PRINTING OFFICE: 1992—336-007

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